NASA CR71510

VOYAGER SPACECRAFT SYSTEM

FINAL TECHNICAL REPORT

TASK B

VOLUME C

ALTERNATE DESIGNS CONSIDERED FOR SPACECRAFT PROPULSION SYSTEMS

D2-82709-8

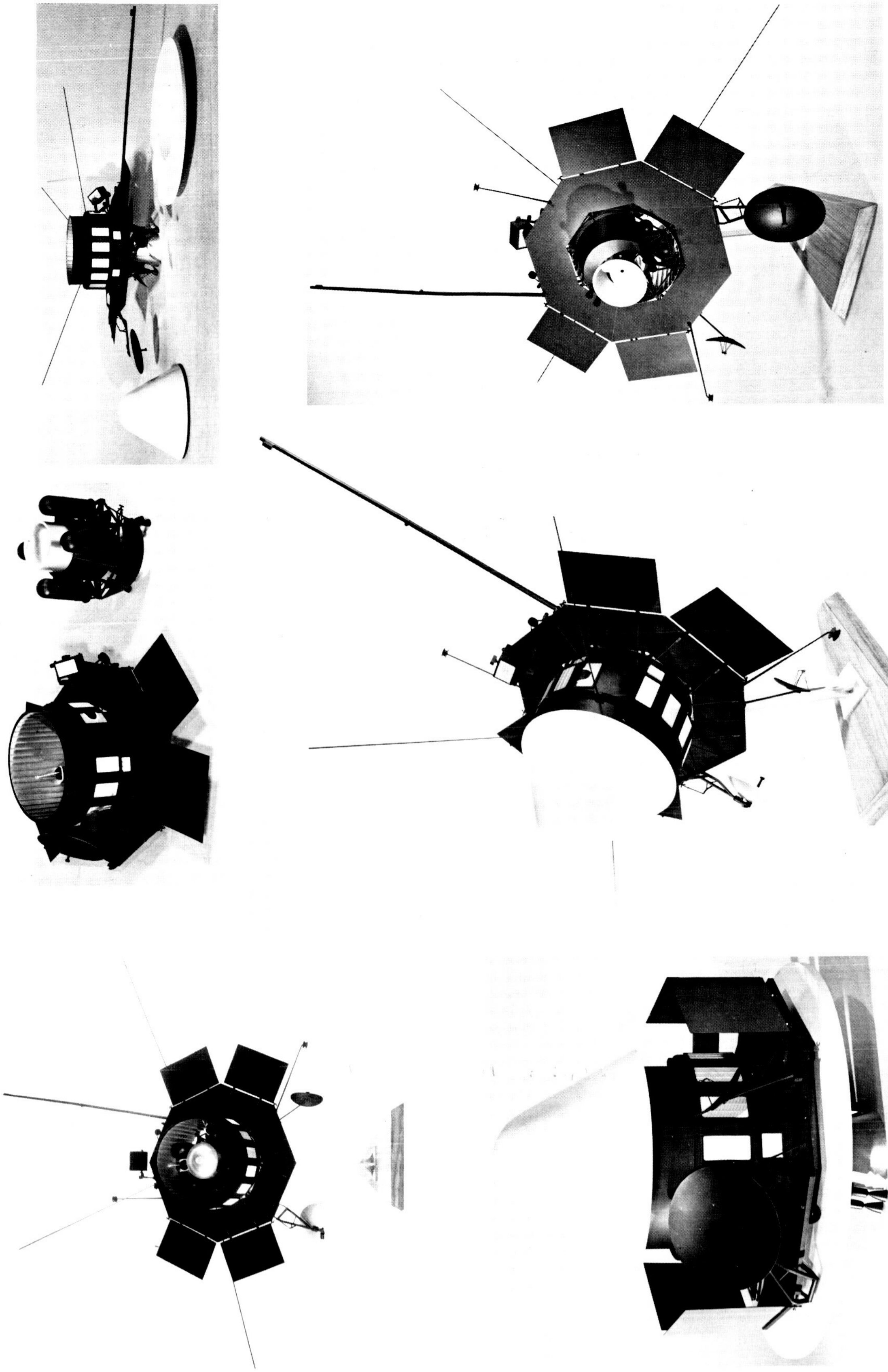
prepared for

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

N66-21075

UNDER
CONTRACT NO. 951111
JANUARY 1966
under NAS 7-100

THE BOEING COMPANY • SPACE DIVISION • SEATTLE, WASHINGTON



Voyager Spacecraft 1/10 Scale Mockup

THE **BOEING** COMPANY

AEROSPACE GROUP • P. O. BOX 3707 • SEATTLE, WASHINGTON 98124

January 21, 1966

IN REPLY REFER TO

Jet Propulsion Laboratory
California Institute of Technology
4800 Oak Grove Drive
Pasadena, California

Gentlemen:

The Boeing Company is pleased to submit the technical reports of the work accomplished under Voyager Phase 1A, Task B. Together with the reports of Task A, they represent to us a substantial contribution to our understanding of the objectives of the Voyager Project. As a corollary, it is believed they will demonstrate to you a dedication for, and a capability to perform, those tasks so important to fulfilling the Spacecraft Contractor's responsibilities.

The recently announced delay in the Voyager Program will test the dedication of all parties concerned. Despite our disappointment, we will not let this temporary setback deter our proceeding on a rational basis to be ready when funding levels again allow the program to proceed. It is important to note that the Task B documentation has been submitted as if no change had occurred in the Voyager Program. It should be recognized that corporate and group commitments contained in the documentation, in the areas of facilities and personnel, will be reconsidered when the Voyager program proceeds. At that time, Boeing will update and reaffirm the resources necessary to support the Voyager program.

Because of the cancellation of the Phase 1B, Part 2 Request for Proposal, we have chosen to highlight some of our management philosophy and organization rationale in a summary document, D2-82709-00. To place this in perspective, we have also summarized the salient features of the spacecraft design. Further, we have postulated some advanced missions, using the 1971 design, for further exploration of the solar system. This latter item is the basis for part of our continuing Voyager work.

Little more remains to be said except to restate that the Voyager Spacecraft System represents to us, more than a new product objective; it is an opportunity to participate in the extension of scientific knowledge in the universe and to contribute to national prestige.

Very truly yours,

THE BOEING COMPANY

Lysle A. Wood

Lysle A. Wood
Group Vice President-Aerospace

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INTRODUCTION

This document, D2-82709-8 (Volume C), "Alternate Designs Considered for Spacecraft Propulsion Systems" is submitted by The Boeing Company in response to Contract 951111, Phase IA, Task B, dated November 2, 1965. The complete technical report in response to Contract 951111, Phase IA, Task B consists of the following:

VOLUME A D2-82709-6 AND D2-82709-9*	PREFERRED DESIGN FOR FLIGHT SPACECRAFT AND HARDWARE SUBSYSTEMS PART I SECTION 1 - VOYAGER 1971 MISSION OBJECTIVES AND DESIGN CRITERIA SECTION 2 - DESIGN CHARACTERISTICS AND RESTRAINTS SECTION 3 - SYSTEM LEVEL FUNCTIONAL DESCRIPTION PART II SECTION 4 - FUNCTIONAL DESCRIPTION OF SPACECRAFT HARDWARE SUBSYSTEMS SECTION 5 - PROGRAM SCHEDULE AND IMPLEMENTATION PLAN
VOLUME B D2-82709-7	DESIGN FOR THE OPERATIONAL SUPPORT EQUIPMENT
VOLUME C D2-82709-8 AND D2-82709-10*	ALTERNATE DESIGNS CONSIDERED FOR SPACECRAFT PROPULSION SYSTEMS

*CLASSIFIED SUPPLEMENT TO VOLUME A AND C RESPECTIVELY

The highlights of the above documentation and management planning are summarized below.

During the period covered by Contract 951111, Task B, Boeing has revised the preliminary design of the Voyager Spacecraft System in consonance with the statement of work. As part of this effort, Boeing has:

1) Verified and revised the requirements and constraints which are imposed upon the Voyager Spacecraft System by the Voyager 1971 Mission.
2) Reviewed and revised the preliminary Flight Spacecraft design for the Voyager 1971 mission, including the study of alternate designs for the spacecraft propulsion systems.
3) Selected a preferred design which reliably and effectively achieves the objectives of the 1971 mission.
4) Reviewed and revised the functional descriptions for the Flight Spacecraft and for each of its hardware subsystems.
5) Reviewed and revised the preliminary requirements and functional description for the Operational Support Equipment (OSE) necessary to accomplish the 1971 mission.
6) Updated and revised the schedule of the Voyager Implementation Plan.

The Boeing Voyager Spacecraft System organization, shown in Figure I-1, is under the direction of Mr. Edwin G. Czarnecki. Mr. Czarnecki is the single executive responsible to JPL and to Boeing management for the accomplishment of the Voyager Spacecraft Phase IA, Task B work and will direct subsequent phases of the program. He reports directly to Mr. George H. Stoner, Vice President and Space Division General Manager.

Although Boeing has capability in all aspects of the Voyager Program it is planned to extend this capability in depth through association with companies recognized as specialists in technologies critical to Voyager

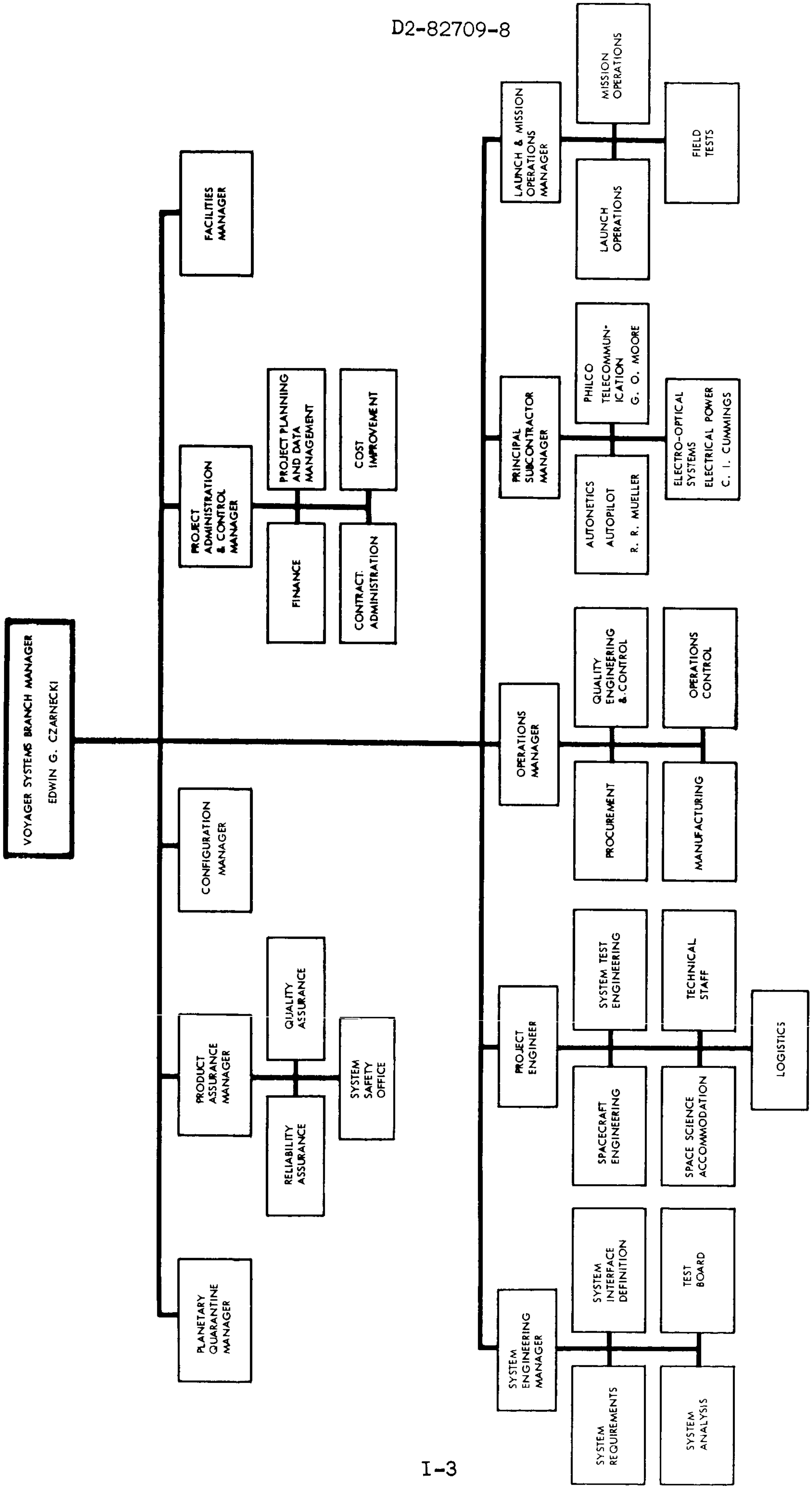


Figure I-1: Boeing Voyager Spacecraft System Organization performance. The following team members have been chosen because of their experience and past performance:

Autonetics, North American Aviation, Anaheim, California

Autopilot and Attitude Reference Subsystem
Mr. R. R. Mueller, Program Manager

Philco, Western Development Lab, Palo Alto, California

Telecommunication Subsystem
Mr. G. C. Moore, Program Manager

Electro-Optical Systems, Inc., Pasadena, California

Electrical Power Subsystem
Mr. C. I. Cummings, Program Manager

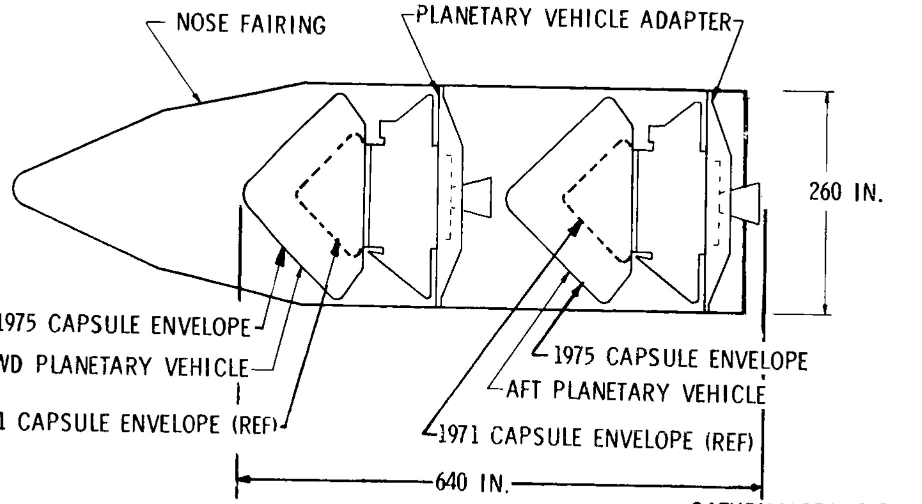
These subcontractor team members have been associated with Boeing on the Voyager Program for periods of 7 to 14 months. As a result of this, there has been sufficient exchange of information to make possible immediate implementation of the project with a Boeing team capable of satisfying the JPL requirements.

The preliminary design approach by the Boeing team has emphasized

- High probability of mission success.
- Conservatism, simplicity, selective redundancy in critical areas, and the use of Mariner experience.
- Versatility to accommodate a wide range of payload, mission, and data requirements.

The Voyager Flight Spacecraft, shown in Figure I-2, has the following principal features:

1) A capability to meet or exceed all mission requirements established in the Voyager 1971 Preliminary Mission Description.

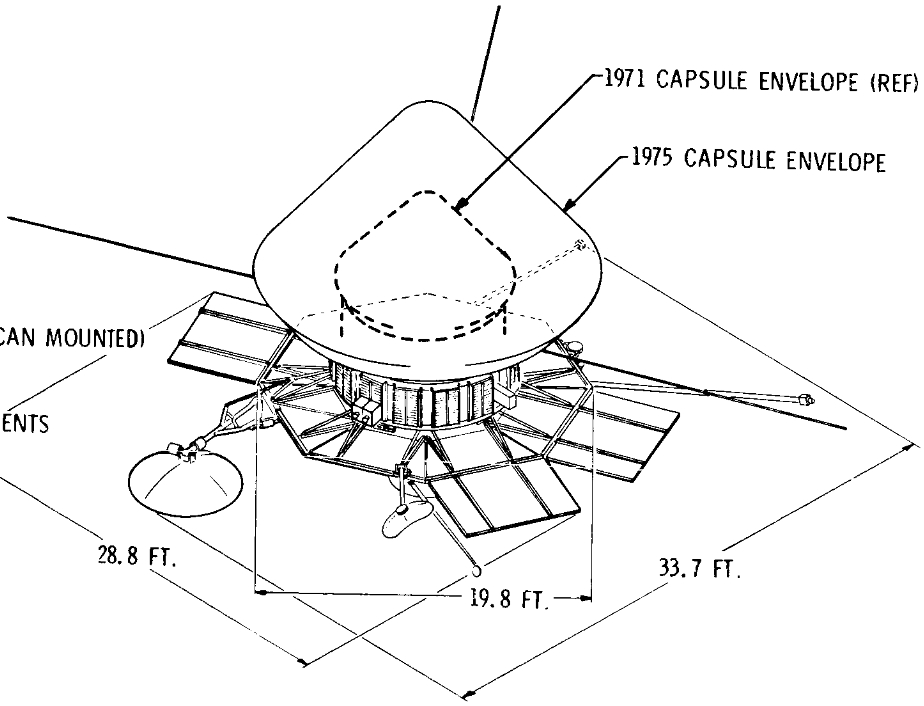


TOTAL SATURN V PAYLOAD ARRANGEMENT

CANDIDATE SPACECRAFT
SCIENCE PAYLOAD

SPACECRAFT BODY - MOUNTED INSTRUMENTS
- PLASMA PROBE
- COSMIC RAY TELESCOPE
- COSMIC DUST DETECTOR
- TRAPPED RADIATION DETECTOR
- ION CHAMBER
- MAGNETOMETER
- RF NOISE DETECTOR
- IONOSPHERE SOUNDER
- BISTATIC RADAR
- GAMMA RAY
- GRAVIMETER
- ULTRAVIOLET SPECTROMETER (SCAN MOUNTED)

SCAN PLATFORM - MOUNTED INSTRUMENTS
- INFRARED SPECTROMETER
- INFRARED SCANNER
- PHOTOIMAGING DEVICE



ASSEMBLED VIEW OF PLANETARY VEHICLE

EXPLODED VIEW OF VOYAGER FLIGHT SPACECRAFT & ADAPTER

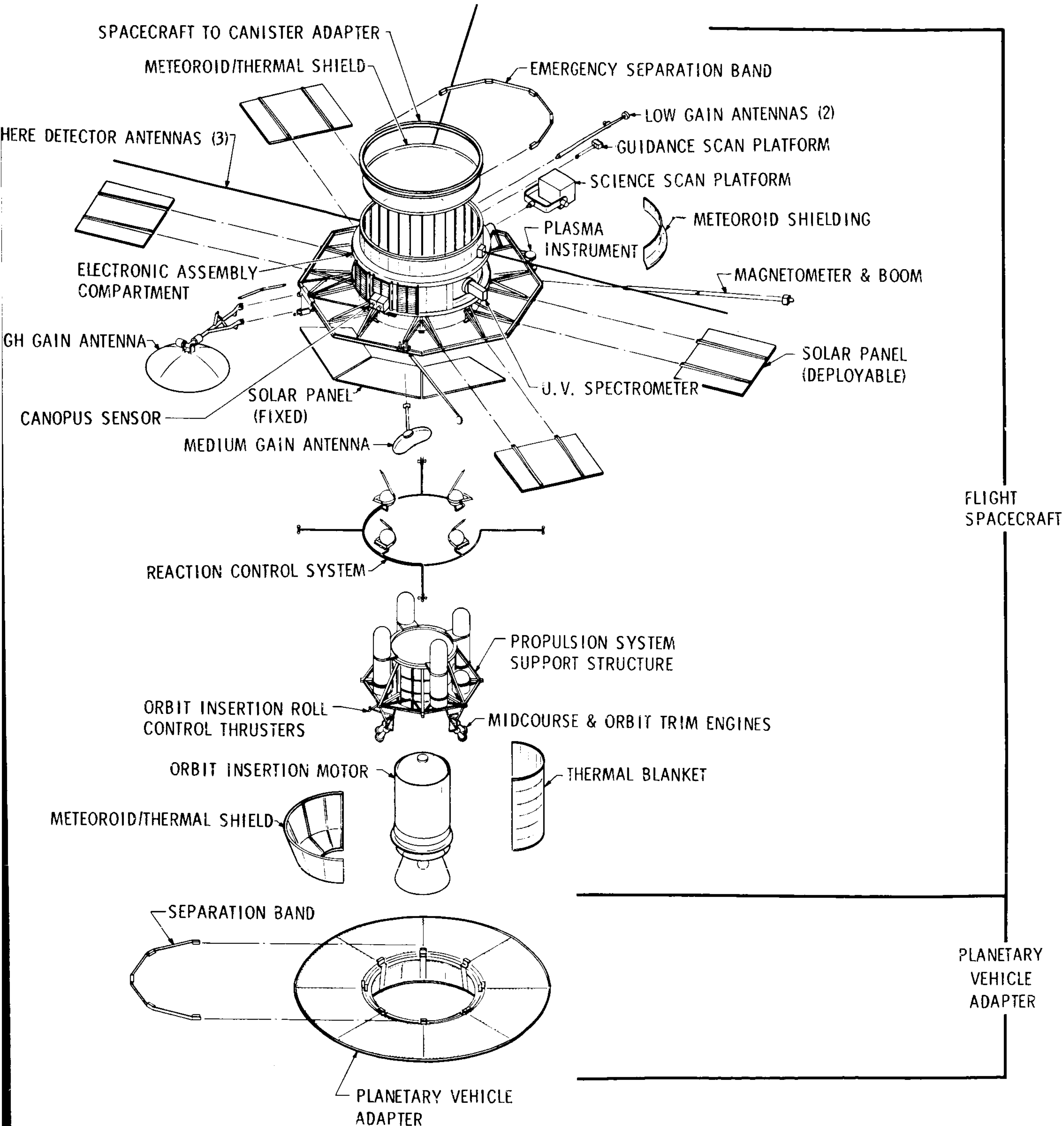


Figure I-2: Voyager Mars Mission Configuration

2) A high probability (approximately 80 percent) of returning science data from at least one spacecraft in Mars orbit. The reliability of the Spacecraft Bus improved from .82 in Task A to .90 in Task B, primarily because of additional redundancy in the telecommunications system.

3) A spacecraft with subsystems sized to accommodate the range of anticipated Mars missions. The 1971 mission capability includes a 93-day launch period, periapsis altitudes as low as 400 km, orbit periods as low as 2.8 hours, and solar occultations as low as 3.7 hours.

4) A single propulsion module capable of fulfilling all Mars mission propulsion requirements from 1971 through 1977 without resizing or changing the propellant quantity.

5) Electrical and electronic systems designed so that no single failure will cause a catastrophic effect on the mission.

6) A computer and sequencer designed so that completion of a nominal mission can be accomplished with programs stored on-board and without ground command intervention unless required by trajectory dispersions or biasing. The ground system can override and back up these programs and command midcourse and orbit corrections when necessary.

7) Space is provided for 16 standard equipment assembly packages, 16" x 32" x 8.5", fastened to the 10-foot-diameter cylindrical structure and thermally interconnected. Fourteen of these are used in the preferred design, all of which employ standardized internal packaging. Thermal control of these assemblies is by space-facing plates radiating through Mariner C type bi-metallic-actuated louvers.

The Flight Spacecraft is 28 feet 10 inches wide, solar panel tip to solar panel tip. The height is 158" compared to a maximum allowable of 208 inches. The estimated weight is 1920 pounds for the Spacecraft Bus. A contingency of 180 pounds is therefore available for selective use during the detail design phase. The estimated weight of the propulsion module is 14,840 pounds with a contingency of 160 pounds (approximately 10 percent of the inert weights) available for use during the design phase.

Analyses and tradeoffs of the four specified Flight Spacecraft propulsion systems indicated that they were nearly equivalent in meeting the JPL specified requirements. The propulsion system selected is the modified Minuteman Wing VI second stage motor for orbit insertion and a hydrazine subsystem using four 200-pound thrust engines for trajectory corrections, and for orbit trim and vernier. The choice of this selected system was based primarily on the lower technical risk in the development of this system and the larger weight available for reallocation. In addition, it makes maximum use of available proven hardware.

A trade study was conducted between propulsion systems sized for 1971, 1973, and 1975, 1977. The study showed that there were only minor differences and that a single design can be developed, tested, and used without change for all missions, 1971 through 1977.

Wide variations in mission requirements are accommodated by the combined use of the solid motor augmented by the hydrazine system for orbit vernier. The performance of the selected propulsion unit exceeds all 1971 mission specification requirements. It provides an orbit insertion velocity increment in 1971 of 2.39 km/sec (2.2 km/sec design goal) with the 2000-pound capsule.

The hydrazine engines selected for trajectory and orbit correction maneuvers utilize a Shell 405 spontaneous catalyst. The engines are of the same type as those selected during the Task A preliminary design. They provide a total velocity change capability of 637 meters per second for the 1971 mission. The hydrazine subsystem has an engine-out capability without malfunction detection and switching. This is accomplished by canting the engines and using jet vane thrust vector controls to maintain the thrust vectors through the vehicle center of gravity. This, together with the use of proven components, results in a high confidence in the predicted reliability of 0.9960 for the preferred propulsion module.

The telecommunications subsystem is sized to meet the mission design requirements. It can accommodate higher data rates, and allow additional modes if such needs develop. The system selected uses a 50-watt traveling wave tube amplifier and a 6-1/2 foot diameter paraboloidal high-gain antenna with two axes of rotation. Complete coverage of Earth is provided during Earth-to-Mars transit, orbit insertion, and orbiting flight. Space is available for growth to an 8 x 12 foot paraboloid. A maximum data rate of 7500 bps is provided with the $6\frac{1}{2}$ foot diameter antenna. The system has the potential for a data rate of 15,000 bps for a period of 20 days after encounter under worst case conditions. A 1260 bps backup mode is available during the first 100 days of Mars orbit. This is accomplished with a fixed Mariner C paraboloidal antenna oriented to provide coverage of Earth during that period.

Five telemetry modes have been provided with data rates of 7500 bps for orbital use, 1260 bps for backup and late mission use, 80 bps for launch and interplanetary cruise, 1.64 bps for emergency use with the low-gain antenna, and an acquisition mode without data transmittal.

Data storage capacity is 3.8×10^8 bits in seven tape recorders. Recording and playback rates can be controlled redundantly through the Data Automation Equipment, Earth Command and the Computing and Sequencing Subsystem.

The Command Subsystem provides for two hundred (27-bit) stored and direct commands with growth provided for by expansion of the output combiner. Two complete, parallel command detectors and decoders with selection logic permits either detector to operate with either decoder to provide high reliability. The probability of executing a false command is several orders of magnitude less than the JPL requirement of 10^{-8}.

The Computing and Sequencing Subsystem controls the sequencing of time-dependent events during the Voyager mission. All functions for a nominal mission can be sequenced from launch through the end of orbital operations without command from mission control unless required by trajectory dispersions or biasing. The selected subsystem is a special-purpose programmable digital computer with an overall reliability of 0.986. It has a capacity for storage of 1024 (27-bit) words and a capability to execute 140 difference commands. Seven-hundred words of storage are required to perform mission functions leaving a 30-percent reserve capacity in a standard size core memory assembly.

The Guidance and Control Subsystem is similar to that selected in Task A and draws heavily on Mariner and Ranger concepts. The Canopus and Sun sensors, the analog type autopilot, and the cold nitrogen reaction control system maintain cruise attitude within ±0.3 degree. A planet sensor, limb detector, and terminator detector have been added to the Task A system. Single-axis ball and air bearing gyros and free rotor gas bearing gyros were re-examined. The free-rotor, gas-bearing Minuteman G6B gyro, modified to a higher torquing capability, was selected because of (1) demonstrated performance in the Minuteman application, and (2) a minimum number of units required for operational redundancy.

Reaction Control is by expulsion of cold nitrogen gas through coupled 0.125 pound pitch and yaw thrusters and coupled .035 pound roll thrusters. Sixteen separate thrusters are provided in a redundant configuration. Four titanium tanks contain 44 pounds of nitrogen. Under nominal conditions the nitrogen will last four years.

The Electrical Power Subsystem has been revised from the Task A design to satisfy new mission and physical constraints. Fixed and deployable panels were evaluated extensively. The selected solar panel array consists of 8 fixed trapezoidal panels (178 square feet), and 4 deployable rectangular panels (138 square feet) for a total of 316 square feet. This configuration meets power requirements for all mission periods and orbit selections, and in addition will meet major mission objectives if one panel fails to deploy. The solar electrical system provides 908 watts of gross power for spacecraft, capsule, and battery charging loads at the end of six months of orbital operation. The configuration can be tested in the Boeing Space Chamber with panels deployed.

Three silver cadmium batteries rated at a total of 2720 watt hours provide power for off-Sun periods up to 3.7 hours. Battery size and circuit design allow the mission to be completed if any one battery section fails. Prime power is distributed at 2400 cycles per second, single phase, 50 volts. Three sets of regulators, inverters, and switching equipment are provided in a redundant configuration. This provides capability to operate all vehicle subsystems in event of a failure of any one power channel. Redundant 400 cycle per second inverters are provided for scan platform controls. Redundant precision oscillators are also provided.

The spacecraft structural arrangement is extensively revised from the Task A preferred configuration because of the larger and heavier propulsion module and increased capsule attachment diameter. Structural weight is 385 pounds and consists of (1) the primary structure assembly; (2) the external supports for appendages; (3) the capsule support and emergency separation assembly; and (4) an eight point Planetary Vehicle separation assembly. The primary structure is a 120-inch diameter magnesium shell, 85 inches long, of conventional semi-monocoque design. This shell provides direct support for attachment of 16 equipment modules (14 used) and for distribution of thermal loads between the assemblies. The Planetary Vehicle adapter is designed to support the spacecraft at eight points and provides uniform loading at the nose fairing interface.

The mechanisms employed for release, deployment, and latching of deployed booms or linkages are the same as those proposed during Task A. Dual bridge-wire, pyrotechnic pin-pullers are used to release the pins holding the various components in their boost positions. Vinson-type actuators were selected for the deployment function; and spring-actuated, taper-pins are used to lock the components in their deployed positions. Self-aligning, spherical bearings are used for all hinge joints to counter any binding effects caused by thermal distortion; and sleeve bearings within the spherical bearings provide a second path of rotation, thus increasing the reliability of the system.

Four-segment, V-block separation bands are used to release the Planetary Vehicle from its adapter and also to effect emergency release of the Flight Capsule. Four pyrotechnic separation devices in each band assure a release reliability of .99992. Eight helical compression springs impart a total separation velocity of 1 foot per second.

The selected pyrotechnic subsystem follows the basic concept of the Mariner series in using capacitors and solid state switches. The pyrotechnic subsystem provides for a set of 21 command signals and 59 electro-explosive devices.

The Temperature Control Subsystem maintains the Spacecraft Bus, propulsion module and science instruments within specified operating temperatures throughout all the mission phases. The design approach, parts, and materials are similar to those used on Mariner C. The equipment modules are coupled thermally and temperature control is accomplished by 52 square feet of bi-metal actuated louvers and high emittance radiator surfaces. The thermal dissipation capacity of the system is approximately 1200 watts, providing nearly 50 percent more capability than required to maintain gross thermal balance.

A science scan platform (GFE) is postulated to support the following science equipment: Infrared Spectrometer, Infrared Scanner, and two television cameras. This platform, with two-axis gimbal drive, provides the science instruments with clear views of Mars. An Ultraviolet Spectrometer is mounted on the spacecraft body with adequate scanning capability.

Substantial additional study and analysis has been made of ways to meet the planetary quarantine requirements and of the resulting Flight Spacecraft design constraints. New data made available or developed since the Task A report are:

1) The new Martian atmosphere which affects both probability of capture and heating rate of contaminated ejecta.
2) Micro-organism IR emissivity which has been determined by Boeing to be approximately 0.2 instead of the previously estimated value of 0.5 to 1.0.
3) Increased microbial kill due to low ultraviolet attenuation in the Martian atmosphere.
4) Reduction by a factor of 10^4 in the meteoroid environment at Mars and associated reduction in the amount of contaminated material spalled off the orbiting spacecraft.
5) Tests run by Boeing which demonstrate with a high confidence that hydrazine is self-sterilizing.
6) Firings of solid engines by Boeing which indicate that the micro-organisms do not survive the hot firing.

Based upon the above, the approaches taken in each hardware area for the selected design are:

1) Micrometeoroid Ejecta--No surface sterilization is provided for the spacecraft, but study and analysis should be continued. The higher ultraviolet kill and the lower micrometeoroid environment reduces the probability of contaminating the planet to 2.8×10^{-5}.

2) Reaction Control and Thrust Vector Control Ejecta, Midcourse and Orbit Trim Pressurant--Sterilize or decontaminate the nitrogen, Freon, and hardware internal surfaces. Study further to assess ultraviolet kill.

3) Midcourse and Orbit Trim Engine--No sterilization of the propellant or propellant hardware in this system is provided because of hydrazine's self-sterilizing characteristics. Tests in Phase IB are required to verify that micro-organisms are not ejected from down stream hardware in Mars orbit.

4) Orbit Insertion Engine--Based upon UV kill and hot firing indications, this engine is not sterilized. Further analysis and hot test firings in Phase IB are required to confirm data prior to initiation of engine procurement.

The OSE selected is a modest extension of Mariner concepts. Subsystem test sets are used as the basic building blocks for the System Test Complex. The System Test Complex employs a Scientific Data Systems general purpose digital computer in a Central Data and Control System for automatic control of the subsystem test sets and central data analysis and display. The total design emphasizes minimum new development to enhance mission success and cost effectiveness.

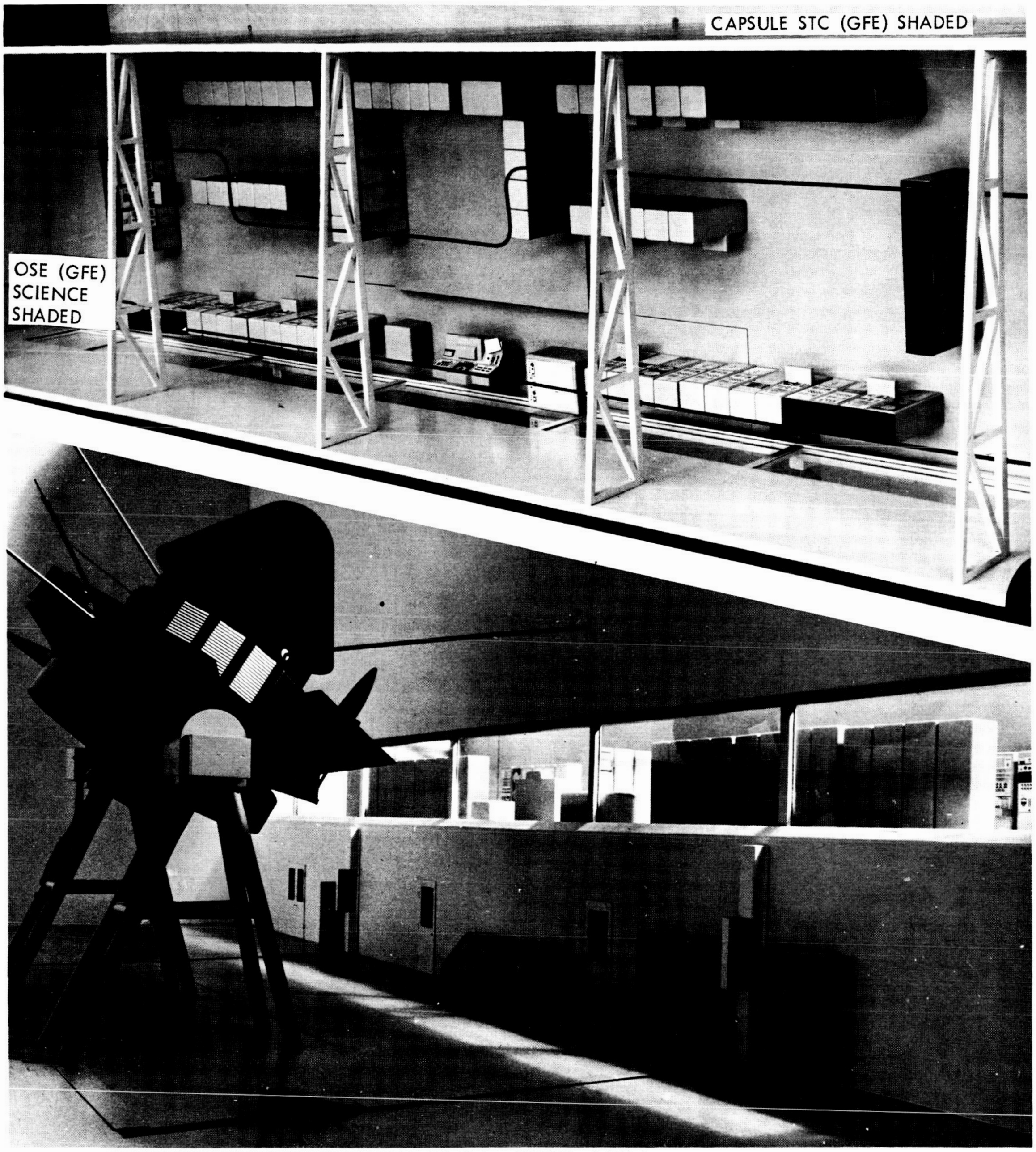
Several existing test systems were considered for use in System Test Complex design and traded off against the preferred concept which is an updated version of that proposed by Boeing in the Phase IA Task A submittal. Systems considered include the Apollo Acceptance Checkout Equipment (ACE) and the Mariner C test equipment. The trade studies indicate that use of ACE would be either non-responsive to specification requirements or, if subsystem OSE were incorporated into the system, would be unnecessarily complex. Mariner C equipment does not include the required degree of central control and automaticity.

All requirements can be met with the preferred design which is well within current technology. It is planned that existing hardware be employed to a maximum degree in defining the Spacecraft System OSE and common components be employed wherever feasible.

The building block approach to design has also been applied to computer program development. Mission operations and test programs are assembled from sub-routines prepared in standard format in accordance with standardized software requirements. This minimizes software development time and costs and allows computer program preparation in parallel with equipment design.

Subsystem Test Sets are typically 1 to 9 standard racks containing equipment similar to that used in the Mariner Subsystem OSE. When elements of these are integrated with the SDS 920 (or 930) computer and appropriate interface adapters, they form a System Test Complex (STC) of approximately 55 cabinets (racks, output data units, and control consoles). Addition of the Mars surface lander capsule and Science Subsystem OSE brings the total Planetary Vehicle System Complex (STC) to about 76 cabinets. Figure I-3 shows a model of the

VIEW LOOKING DOWN



VIEW FROM TEST AREA

Figure I-3: System Test Complex And Equipment

STC, typical test facilities, and equipment. Elements of the STC are employed as an integral part of Launch Complex Equipment (LCE).

A test team concept is planned wherein technical personnel experienced in spacecraft and OSE design, systems test operations, launch and mission operations, and spacecraft assembly and quality control will be formed into test groups. One of these teams will be assigned to each flight spacecraft and spare and will follow that vehicle from assembly through launch. Selected elements of the test team will continue to support mission operations for their spacecraft.

The Task B review and revision of the preliminary design for the Voyager Spacecraft System has emphasized conservative design, particularly in the use of proven equipment and techniques to the greatest extent consistent with system requirements. High reliability has been achieved through selection of space-proven components and through design of redundant capabilities into subsystems and equipment. The propulsion subsystem has been sized to achieve a range of flight trajectories and Mars orbits for missions in the years 1971 through 1977. The preferred Flight Spacecraft design provides mission versatility and capability for growth. As a result of the Task B activities, The Boeing Company has developed a design believed to be optimum for achieving objectives of the Voyager 1971 mission.

SUMMARY

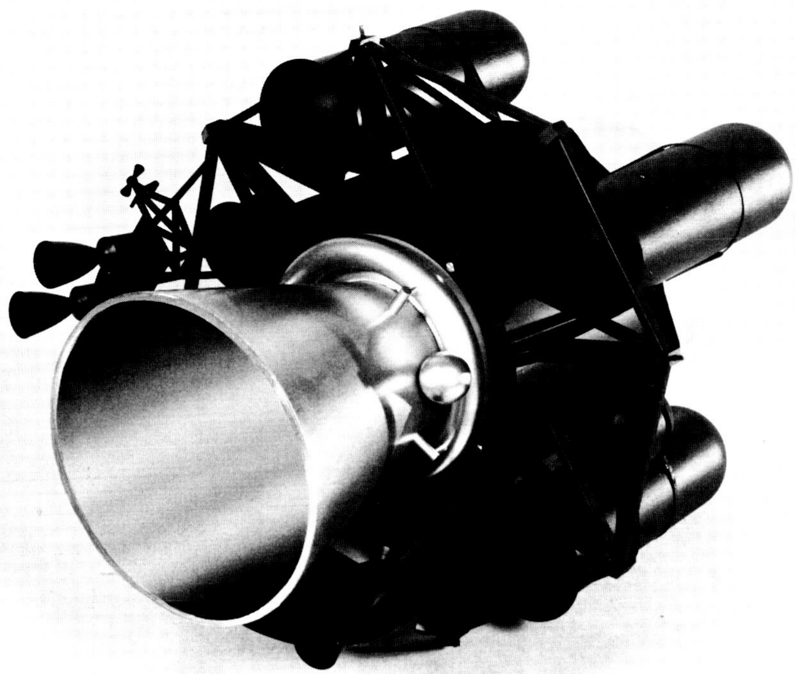
Preferred-Design Description

The preferred spacecraft propulsion module, shown in Figure S-1, consists of a solid motor for orbit insertion and a variable-impulse, multiple-start, liquid-monopropellant subsystem for midcourse corrections, orbit trim, and orbit insertion vernier. The module, which weighs 15,000 pounds, is sized to accommodate a 3500-pound Spacecraft Bus and a 10,000-pound Flight Capsule. It satisfies all Voyager Mars propulsion design goals from 1971 through 1977 without resizing. The predicted reliability of the preferred propulsion system for the Voyager Mars mission is 0.996.

The solid unit is a modified Minuteman Wing VI second-stage motor. Modifications consisted of the following:

1) Shortening the motor case by 36 inches;
2) Extending the nozzle skirt by 15 inches;
3) Changing freon tank material to titanium, freon pressurant to nitrogen, and deleting hydraulic servo pumps and hydraulic fluid.

The preferred-design titanium motor case is cylindrical with elliptic fore and aft domes. Overall motor length from safe and arm to nozzle exit plane is 144.3 inches. The partially buried nozzle has an exit-to-throat area ratio of 32.5. The propellant contains 15-percent aluminized polybutadiene cast in a finocyl grain. Regressive burning results in a 3.5-g maximum acceleration during a 1971 orbit insertion with a 2000-pound Flight Capsule. Ignition is provided by a forward-mounted sealed pyrogen igniter. Pitch and yaw thrust vector control is by electro-hydraulic secondary injection of Freon 114B2. Maximum effective thrust



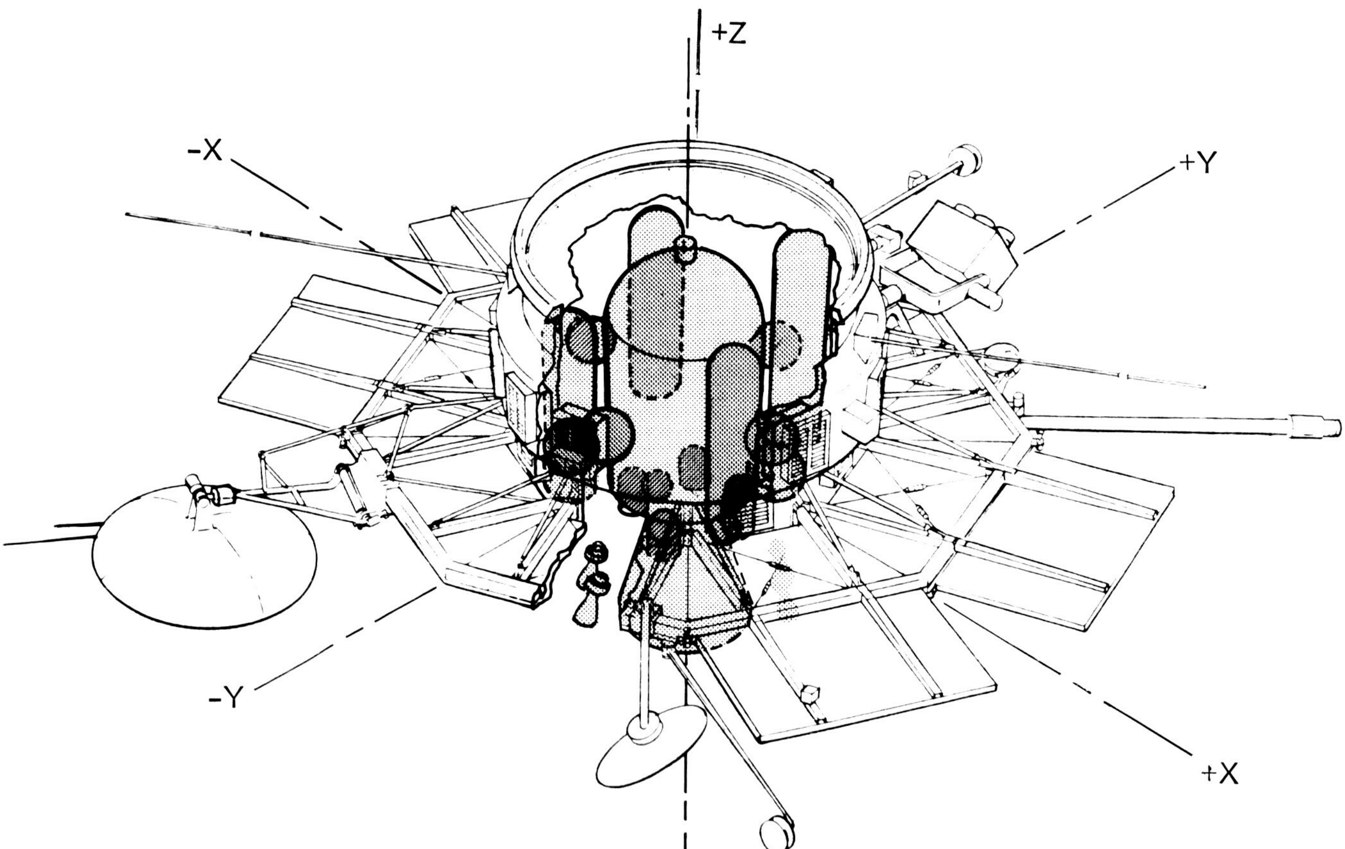


Figure S-1: Preferred Propulsion System

vector angular deflection required is ±2 degrees. The Freon is stored in a titanium tank, is expelled by a viton rubber bladder, and uses regulated nitrogen gas as a pressurant. Roll thrust vector control is provided by 6-pound nitrogen thrusters with quad redundant valves. The solid motor, including its thrust vector and roll-control system, weighs 10,400 pounds and results in a propulsion module orbit insertion velocity increment capability of 2.39 Km/sec (7833 fps) to a 1971 Planetary Vehicle with a 2000-pound Flight Capsule. The surfaces of the solid-motor thrust vector control assembly components exposed to nitrogen and Freon will be decontaminated by ethylene oxide to comply with planetary quarantine constraints. Filtered nitrogen and Freon will be loaded aseptically.

The liquid monopropellant subsystem uses four 200-pound thrust, regulated pressure-fed, radiation-cooled hydrazine engines. Multi-start capability is provided by the Shell 405 spontaneous-decomposition catalyst. The hydrazine engines are mounted well aft and are canted at 13 degrees to the roll axis. Thrust vector control is accomplished by jet vanes. Maximum effective thrust vector angular deflection is ±5 degrees. The subsystem has an engine-out capability without recourse to malfunction detection and switching. A total of 3190 pounds of usable hydrazine is stored in four cylindrical tanks with butyl bladders for positive expulsion. Eighty-nine pounds of regulated nitrogen gas, stored in two spherical tanks, provided for propellant tank pressurization. All hydrazine system valving is redundant for increased reliability. Positive isolation of both pressurant and propellant is provided up to the second midcourse correction maneuver to minimize leakage.

Surfaces of the liquid monopropellant pressurization system exposed to nitrogen will be decontaminated by ethylene oxide and loaded aseptically with filtered nitrogen to comply with planetary quarantine requirements.

The preferred propulsion design can insert the planetary vehicle into all specified orbits whose velocity increment requirements do not exceed the maximum total impulse capability of the system. This is accomplished by: 1) Inserting into an intermediate orbit with the solid motor, and 2) Providing an orbit insertion vernier with the hydrazine subsystem. The preferred Voyager propulsion subsystem design is described in greater detail in Volume A, Section 4.3.

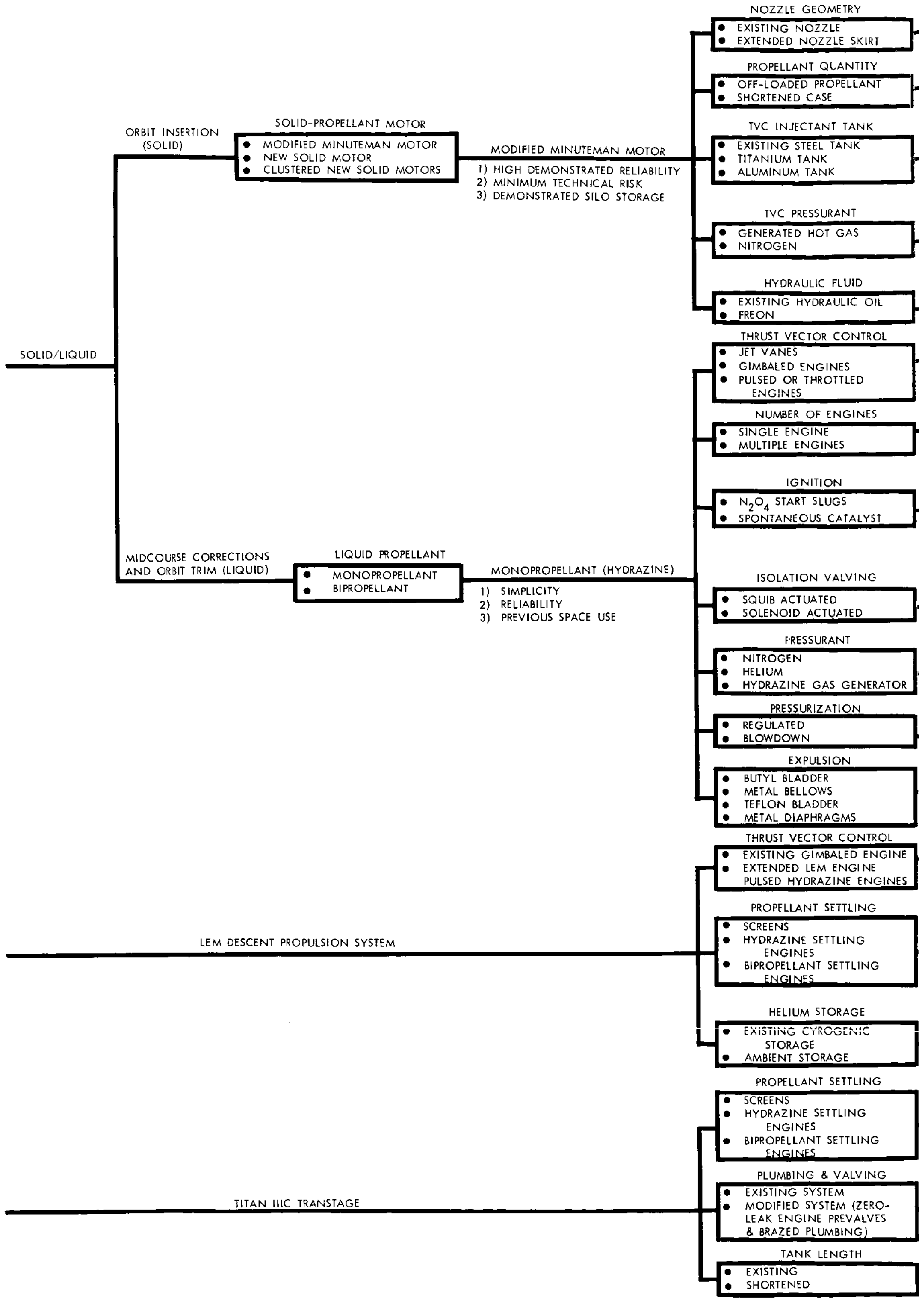
Candidate Concepts

The four propulsion systems considered were: 1) A solid/liquid system sized for the 1971 and 1973 missions, 2) A solid/liquid system sized for the 1975 and 1977 missions, 3) The Apollo Lunar Excursion Module (LEM) descent propulsion system, and 4) the Titan III-C transtage. Selection of the preferred propulsion design followed the logic tree that is shown in Figure S-2.

In sizing the solid/liquid units, three solid-motor designs were considered:

1) A modified Minuteman Wing VI second-stage motor;
2) A new solid motor;
3) A cluster of new solid motors.

The modified Minuteman Wing VI second-stage motor was selected as the preferred solid-motor design. This concept represents a minimum technical risk in view of the high demonstrated reliability of the existing Minuteman motor. A new solid-motor design offers significant velocity performance gains over the modified Minuteman motor, and could demonstrate high reliability with sufficient testing. A decision in



SOLID/LIQUID
ORBIT INSERTION (SOLID)
SOLID-PROPELLANT MOTOR
• MODIFIED MINUTEMAN MOTOR
• NEW SOLID MOTOR
• CLUSTERED NEW SOLID MOTORS
MODIFIED MINUTEMAN MOTOR
1) HIGH DEMONSTRATED RELIABILITY
2) MINIMUM TECHNICAL RISK
3) DEMONSTRATED SILO STORAGE
NOZZLE GEOMETRY
• EXISTING NOZZLE
• EXTENDED NOZZLE SKIRT
PROPELLANT QUANTITY
• OFF-LOADED PROPELLANT
• SHORTENED CASE
TVC INJECTANT TANK
• EXISTING STEEL TANK
• TITANIUM TANK
• ALUMINUM TANK
TVC PRESSURANT
• GENERATED HOT GAS
• NITROGEN
HYDRAULIC FLUID
• EXISTING HYDRAULIC OIL
• FREON
MIDCOURSE CORRECTIONS AND ORBIT TRIM (LIQUID)
LIQUID PROPELLANT
• MONOPROPELLANT
• BIPROPELLANT
MONOPROPELLANT (HYDRAZINE)
1) SIMPLICITY
2) RELIABILITY
3) PREVIOUS SPACE USE
THRUST VECTOR CONTROL
• JET VANES
• GIMBALED ENGINES
• PULSED OR THROTTLED ENGINES
NUMBER OF ENGINES
• SINGLE ENGINE
• MULTIPLE ENGINES
IGNITION
• N_2O_4 START SLUGS
• SPONTANEOUS CATALYST
ISOLATION VALVING
• SQUIB ACTUATED
• SOLENOID ACTUATED
PRESSURANT
• NITROGEN
• HELIUM
• HYDRAZINE GAS GENERATOR
PRESSURIZATION
• REGULATED
• BLOWDOWN
EXPULSION
• BUTYL BLADDER
• METAL BELLOWS
• TEFLON BLADDER
• METAL DIAPHRAGMS
LEM DESCENT PROPULSION SYSTEM
THRUST VECTOR CONTROL
• EXISTING GIMBALED ENGINE
• EXTENDED LEM ENGINE
PULSED HYDRAZINE ENGINES
PROPELLANT SETTLING
• SCREENS
• HYDRAZINE SETTLING ENGINES
• BIPROPELLANT SETTLING ENGINES
HELIUM STORAGE
• EXISTING CYROGENIC STORAGE
• AMBIENT STORAGE
TITAN IIIC TRANSTAGE
PROPELLANT SETTLING
• SCREENS
• HYDRAZINE SETTLING ENGINES
• BIPROPELLANT SETTLING ENGINES
PLUMBING & VALVING
• EXISTING SYSTEM
• MODIFIED SYSTEM (ZERO-LEAK ENGINE PREVALVES & BRAZED PLUMBING)
TANK LENGTH
• EXISTING
• SHORTENED

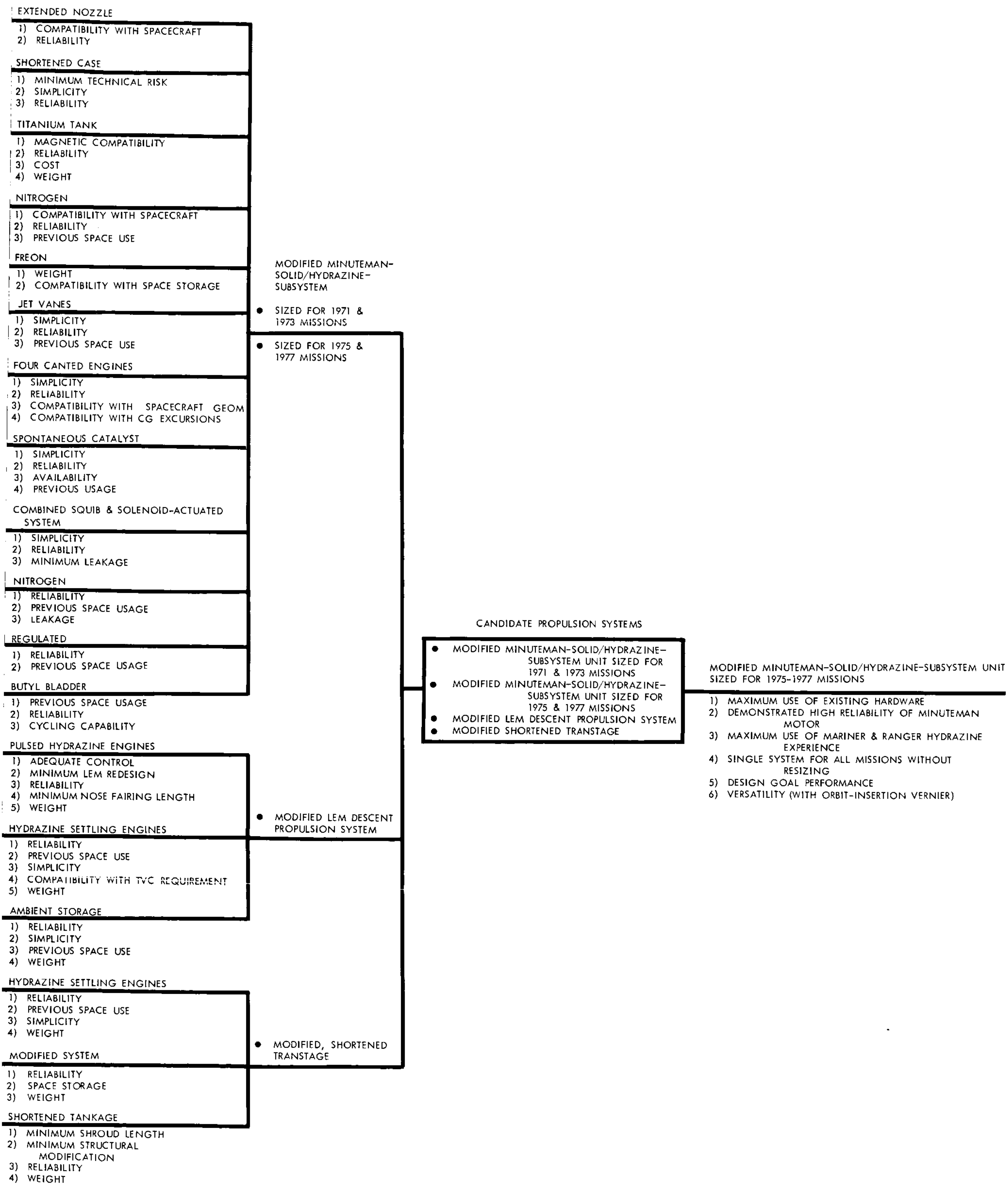


Figure S-2: Propulsion System Selection Logic Chart favor of a new solid motor cannot be made, however, without firm cost data to establish its cost effectiveness. Solid-motor specifications are to be released to qualified propulsion vendors. These specifications can be met by either a modified Minuteman motor or a new solid. The preferred solid-motor design selection will be reviewed with the aid of forthcoming propulsion vendor design, schedules and firm cost data.

The clustered-new-solids concept was rejected since it could not meet--due to high inerts--the minimum 1971 orbit insertion velocity increment of 2.0 Km/sec with a 2000-pound capsule.

In sizing the liquid subsystem of the solid/liquid units, both mono-propellant and bipropellants were considered. A hydrazine monopropellant subsystem was selected because it is a simpler, more reliable, space-proven subsystem.

Adapting the LEM descent propulsion system to Voyager requirements necessitated significant modifications in the following:

1) Thrust vector control;
2) Propellant settling;
3) Pressurant storage.

The existing LEM descent propulsion system uses engine gimbaling for thrust-vector control (TVC). Existing gimbal angle capability is inadequate for Voyager vehicle center-of-gravity locations. A pulsed-hydrazine-engine subsystem was therefore added to provide TVC. It is the only TVC concept feasible without major LEM system redesign. The hydrazine subsystem is also used for settling the propellants for the

LEM descent engine. The existing LEM descent stage relies on cryogenic temperature storage of helium, which is used as propellant pressurant. Because of the long duration of the Voyager mission, ambient temperature high-pressure storage of helium was substituted.

Adapting the Titan III-C transtage to Voyager requirements necessitated significant modifications in the following:

1) Propellant settling;
2) Plumbing and valving;
3) Tank length.

The transtage control module, which includes the propellant settling rockets, is removed for Voyager applications because it contains much unnessary equipment which resulted in high inert weight. A highly reliable four-engine hydrazine subsystem was therefore added to transtage for propellant settling. The main transtage engines were equipped with zero-leak prevalves, and all plumbing connections were brazed to minimize leakage. Because of the desire to reduce booster loads, a shortened transtage was selected over the existing transtage. The propellant tanks were shortened by 20 inches to minimize nose fairing length. Shortening the tank length by more than 20 inches is not feasible without major structure redesign.

Final selection of the preferred design was made from the following candidate systems:

1) A modified Minuteman motor/hydrazine subsystem sized for the 1971 and 1973 missions;

2) A modified Minuteman motor/hydrazine subsystem sized for the 1975 and 1977 missions;

3) A modified LEM descent propulsion system;

4) A modified, shortened transtage.

Competing characteristics for selecting the preferred design, in order of priority, were:

1) Probability of mission success;

2) Performance of mission objectives;

3) Cost savings;

4) Contributions to subsequent missions;

5) Additional 1971 capability.

The modified Minuteman motor/hydrazine subsystem unit, sized for the 1975 and 1977 missions, was selected as the preferred design because: 1) It results in the highest propulsion contribution to probability of mission success by virtue of the high demonstrated reliability of the Minuteman motor and the Mariner IV hydrazine subsystem experience; 2) It provides a single propulsion unit for all Voyager Mars missions through 1977 without resizing; 3) It exceeds the 1971 orbit insertion velocity increment design goal with a 2000-pound capsule; 4) It exceeds the minimum required 1971 orbit insertion velocity increment with a 3000-pound capsule; and 5) It provides as much versatility for attaining all feasible orbit sizes and orientations as a pure liquid stage of equal Δv capability.

The modified Minuteman motor/hydrazine subsystem unit, sized for the 1971 and 1973 missions, provides a higher 1971 orbit insertion velocity increment than the preferred design. It can impart a 2.25 Km/sec

(7395 fps) velocity increment to a 3000-pound capsule against a 2.17 Km/sec (7137 fps) velocity increment for the preferred design. It was rejected because it required system resizing and requalification for 1975 and 1977 missions. This reduces the probability of mission success in 1975 and 1977 and increases overall program cost.

The modified LEM descent propulsion system that was considered offered the attractive features of: 1) Mission versatility by virture of throttling and thrust termination; 2) Shortest configuration under the nose fairing; and 3) Assurance of the development of the unmodified LEM to a demonstrated high reliability before 1971 because of its central role in the Apollo program. The modified LEM descent propulsion system is competitive with the preferred design. It was not selected because it could not offer as high a probability of mission success as the preferred design. The predicted reliability of the modified LEM descent propulsion system for the Voyager mission is 0.990, slightly lower than the 0.996 for the preferred design. This is primarily because of the reliability degradation caused by the pulsed-hydrazine-engines TVC system. Use of the LEM propulsion system entails a higher technical risk, since it is still in development. The compatibility of bipropellant stages, such as LEM descent propulsion, with prolonged space storage is yet to be demonstrated. Another consideration is the close tie-in of LEM with the Apollo program. Design changes during the Apollo-LEM development are possible, and these could adversely affect compatibility with Voyager requirements without Voyager Program control.

The modified, shortened, Titan III-C transtage that was considered for Voyager was competitive for the following reasons: 1) It is a developed system, currently being flight-tested; and 2) It is the only bipropellant stage that has demonstrated short-term storability and multiple-restart capability in space. It was not selected for Voyager as it was slightly inferior to the preferred design in the five key competing characteristics. The transtage, in its current configuration, has a serious leakage problem which significantly degrades its reliability and performance. In its adaptation to the Voyager mission, it was assumed that present leakage rates can be reduced by an order of magnitude. Even so, its predicted reliability (0.991) is less than that of the preferred design.

Preferred Design--Solid-Motor Subsystem, Orbit Insertion

Modifications to the selected Minuteman Wing VI second-stage motor were few to maintain the demonstrated high reliability of the motor. Only those changes that improve probability of mission success and meet mission requirements were adopted. Modifications were examined in the following areas: motor geometry, nozzle geometry, and thrust vector control injectant, pressurant, and assembly.

1) Motor Geometry--The Minuteman motor's propellant loading is significantly in excess of Voyager requirements. Offloading propellant and shortening the motor case were considered. A 36-inch shortened case was selected because it involves a smaller technical risk, is easier to implement, and results in a lower acceleration during motor burn.
2) Nozzle Geometry--The existing nozzle with and without a 15-inch nozzle skirt extension was considered. The nozzle extension increases the nozzle expansion ratio from 24.8 to 32.5. The extended nozzle was selected because it reduces plume radiant heating to the solar panels, thereby increasing the probability of mission success.

3) Thrust Vector Control Injectant and Pressurant--Freon and hydrazine were considered as injectants for pitch and yaw thrust vector control. Freon was selected on the basis of minimum technical risk and is currently in use on the selected motor. Generated hot gas and stored nitrogen were considered for freon pressurization. Generated hot gas is currently in use on the motor. Excess hot gas is regulated by overboard dumping. Nitrogen was selected because it is easier to regulate and provides a more favorable thermal environment. Also, overboard dumping of solid-generated hot gas can be detrimental to optical sensors and surface coatings.

4) Thrust Vector Control Assembly--The existing Minuteman motor TVC assembly, with and without modifications, and a new assembly were considered. A new TVC assembly was rejected. Its slight weight savings and improved control-loop dynamics did not justify the increased technical risk. The existing TVC assembly was not acceptable because the freon stainless-steel tank is ferromagnetic, and the hydraulic servo valves require excessive power. A modified Minuteman motor TVC assembly was therefore selected as the preferred design. The modified assembly includes an identically shaped titanium tank for freon storage, and uses freon as the hydraulic fluid. The present hydraulic valve can operate on freon with minor plumbing changes. The large power-consuming hydraulic pumps on the Minuteman assembly are no longer required. Excess freon is injected into the nozzle after motor depletion. This differs from the overboard dumping technique of the current design.

Preferred Design--Hydrazine Subsystem, Midcourse, Orbit Trim, and Vernier

The selected hydrazine subsystem is conservative. It relies heavily on Ranger and Mariner hydrazine hardware and technology. Monopropellant subsystem modifications to Mariner propulsion were made in the following areas:

1) Ignition--Hypergolic-start slugs and a spontaneous-decomposition catalyst were considered. The spontaneous-decomposition catalyst was selected because it improves reliability. Its use is warranted because of the large amount of available test data.

2) Engines--A single-engine installation was not feasible. A multiple-engine subsystem with engine-out capability (without malfunction detection and switching) was desired without compromising the use of highly reliable, space-proven jet vanes for thrust vector control. A small number of canted engines and a large number of uncanted engines were considered. Four engines, canted at 13 degrees, were selected as the smallest number of engines for which the performance loss due to cant (2.63 percent) is still acceptable. A thrust level of 200 pounds per engine was selected. It is compatible with a previously fabricated hydrazine engine. It also resulted in a near-minimum-weight subsystem for all Voyager Mars missions through 1977.

3) Isolation Valving--Both solenoid-actuated and squib-actuated valves were considered and a combination of solenoid- and squib-actuated valves was selected for increased reliability. This valving system differs from that of Mariner IV and reflects the increased number of Voyager liquid-subsystem starts and more complex flight sequence of Voyager.

1.0 SCOPE

Each Planetary Vehicle requires propulsion capability for midcourse trajectory corrections which remove or reduce trajectory dispersions and provide for trajectory biasing, including the ten-day arrival date separation of two simultaneously launched Planetary Vehicles. Additional propulsion capability is required for inserting the Planetary Vehicle into orbit around Mars and subsequent orbit trim maneuvers.

Candidate concepts considered were solid/liquid systems sized both for the 1971 and 1973 missions and the 1975 and 1977 missions, the Apollo LEM descent propulsion system, and the Titan III-C transtage. This volume presents propulsion-system tradeoffs and analyses leading to a selection of the preferred propulsion design for Voyager Mars 1971 mission. Section 2.0 describes the optimized candidate propulsion systems. Their competing characteristics, leading to the preferred design selection rationale, are given in Section 3.0. Trade studies leading to the optimum candidate propulsion designs are summarized in Sections 4.0 through 7.0. Section 8.0 presents an assessment of the preferred design.

In conducting the tradeoffs, velocity requirements and weight allocations presented in the "Voyager 1971 Preliminary Mission Description" were used. Candidate propulsion designs for the 1971 mission were sized for a 3000-pound capsule, which slightly penalizes the performance of the solid/liquid designs. A solid/liquid system sized for a 3000-pound capsule can accommodate a 2000-pound capsule without redesign. Similarly, candidate solid/liquid designs for the 1975 and 1977 missions were sized for a 10,000-pound capsule.

PART I PREFERRED DESIGN SELECTION

2.0 OPTIMUM CANDIDATE PROPULSION SYSTEMS

Trade studies were conducted to optimize the four candidate propulsion systems concepts for Voyager. The optimized designs are described below.

2.1 OPTIMUM SOLID/LIQUID SYSTEM, SIZED FOR 1971 & 1973 MISSIONS

The optimum solid/liquid propulsion system for the 1971 & 1973 Voyager missions consists of a solid motor for orbit insertion and a mono-propellant-liquid system for midcourse corrections, orbit vernier, and orbit trim requirements. The installation of this propulsion system within the spacecraft bus is shown in Figure 2.1-1. It is comprised of a modified Minuteman Wing VI second-stage solid motor with fluid-injection pitch and yaw thrust vector control. It is mounted in the center of the propulsion module and surrounded by four cylindrical hydrazine tanks, and two spherical gaseous nitrogen pressurant bottles. There are four canted 200-pound-thrust, radiation-cooled, monopropellant engines with jet vane thrust vector control. A schematic of this propulsion system is shown in Figure 2.1-2. A component list is given in Table 2.1-1. The Minuteman motor is modified to reduce propellant weight and increase nozzle expansion ratio. The liquid-injection thrust vector control system is modified to meet the Voyager requirements. The required reduction in propellant weight for Voyager is accomplished by decreasing the length of the cylindrical case by 30 inches. The existing nozzle has been extended 15 inches to an expansion ratio of 32.5 to decrease plume radiosity. The vacuum specific impulse is increased from * * seconds (nozzle exit half angle decreased from 14.5 to 11.5 degrees).

*See D2-82709-10 Classified Supplement - Reference Page 1

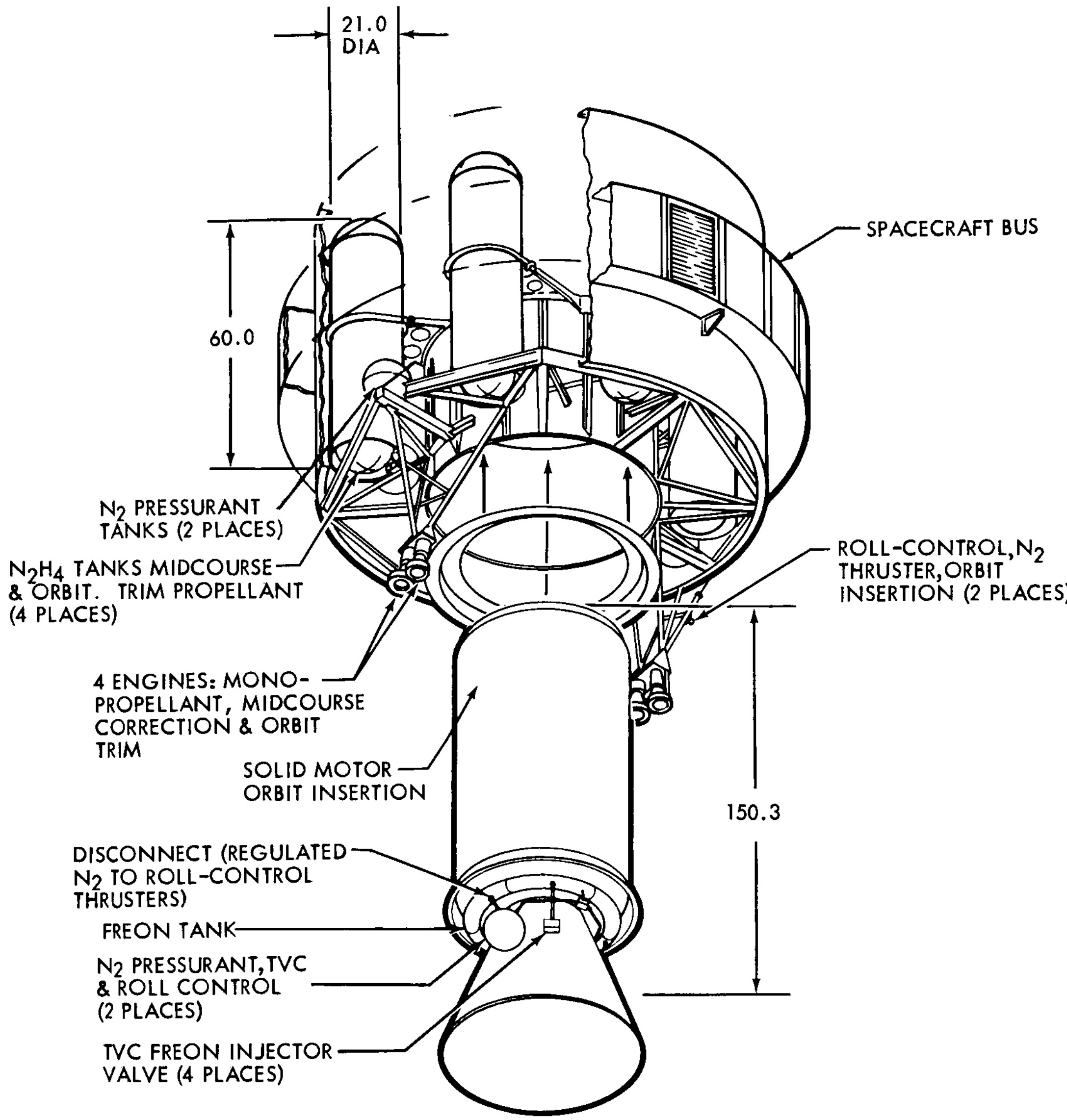


Figure 2.1-1: Solid/Liquid Propulsion System — 1971-1973 Module

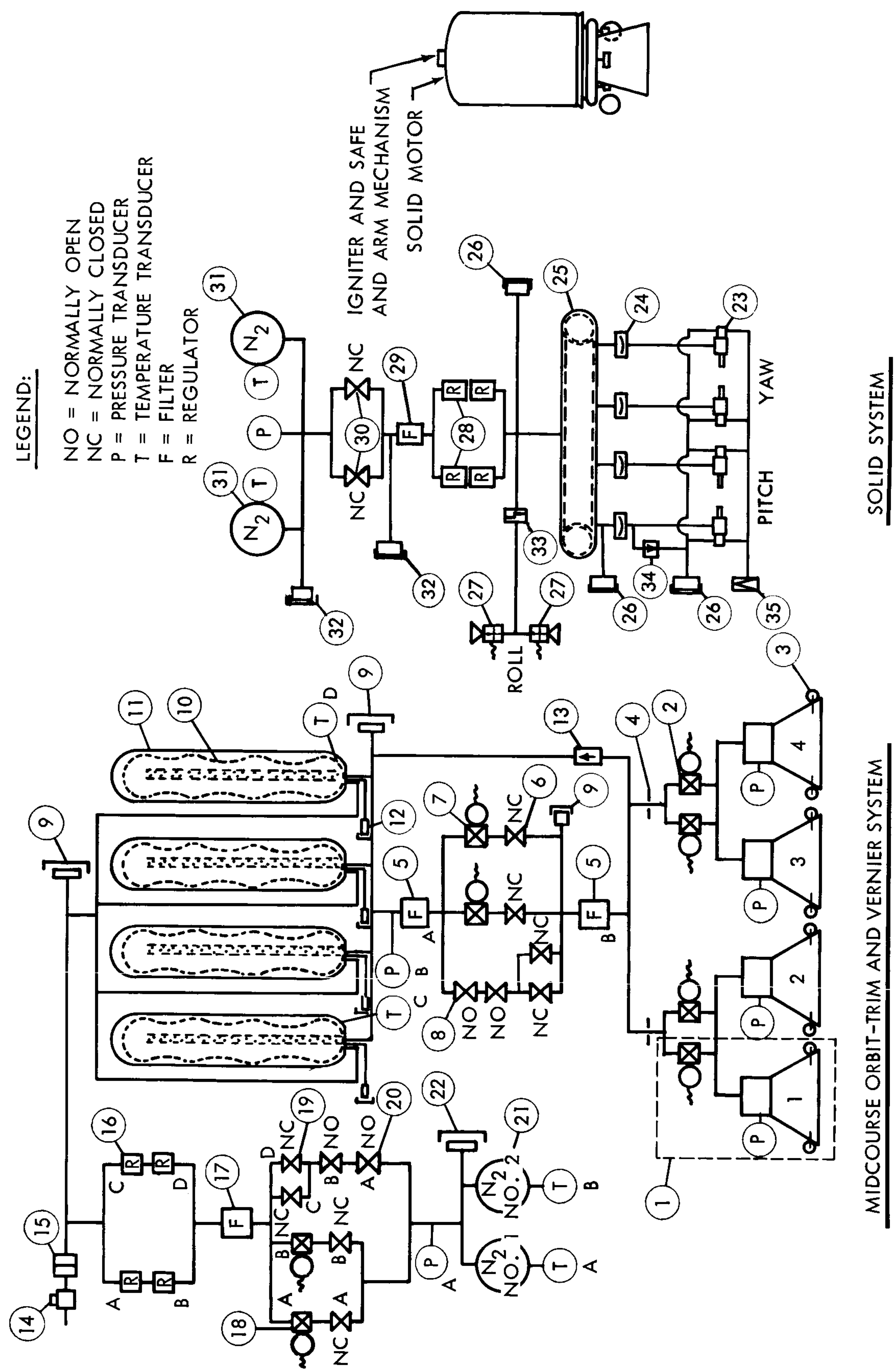


Figure 2.1-2: 1971 — 1973 Solid/Liquid System

Table 2.1-1: 1971 & 1973 SOLID/LIQUID SYSTEM COMPONENT LIST

ITEM	QTY	NAME
		MIDCOURSE, ORBIT TRIM AND VERNIER MONOPROPELLANT SUBSYSTEM
1	4	Rocket Engine Assembly
2	4	Valve, Propellant, Solenoid Latching
3	16	Jet Vane and Actuator Assembly
4	2	Orifice, Propellant Flow
5	2	Filter, Propellant
6	4	Valve, Propellant, Normally Closed, Squib
7	2	Valve, Propellant, Solenoid Latching
8	2	Valve, Propellant, Normally Open, Squib
9	2	Valve, and Cap, Propellant, Fill and Test
10	4	Bladder, Propellant Tank, Expulsion
11	4	Tank, Propellant
12	4	Valve and Cap, Vent, Manual
13	1	Valve, Propellant, Thermal Relief
14	1	Valve, Pressure Relief
15	1	Burst Disk
16	2	Regulator, N_2 Pressure, Dual
17	1	Filter, Nitrogen
18	2	Valve, Nitrogen, Solenoid Latching
19	4	Valve, N_2, Normally Closed, Squib
20	2	Valve, N_2, Normally Open, Squib
21	2	Tank, Nitrogen Pressure
22	1	Valve and Cap, N_2 Fill and Test
		SOLID MOTOR AND TVC
23	4	Servo Valve, Freon Injectant
24	4	Burst Disk and Filter
25	1	Tank, Freon, Bladder Expulsion
26	3	Valve and Cap, Fill and Test
27	2	Valve and Thruster Assembly, Roll Control
28	2	Regulator, N_2 Pressure, Dual
29	1	Filter, Nitrogen
30	2	Valve, N_2 Normally Closed, Squib
31	2	Tank, Nitrogen Pressure
32	1	Valve and Cap, Fill and Test
33	1	Quick Disconnect
34	1	Valve,Check
35	1	Valve,Vent

The four monopropellant engines are canted 13 degrees with respect to the vehicle centerline to provide an engine-out capability. The use of spontaneous catalyst ensures reliable decomposition of N_2H_4 at all temperatures above 0^oF. Four 60 by 20.5-inch cylindrical tanks with positive-expulsion butyl bladders, store 2495 pounds of usable hydrazine. Two 18.25-inch-diameter spherical tanks store 70 pounds of nitrogen gas at 3500 psia for pressurization. Isolation valve assemblies are installed between the nitrogen pressurant reservoir and the propellant tanks, and between the propellant tanks and the engines. These valve assemblies reduce leakage and prevent over-pressurization of propellant tanks and engine inlet valving.

The 1971 and 1973 solid/liquid system performance is presented in Table 2.1-2 for a 3000-pound capsule and Table 2.1-3 for a 2000-pound capsule. For the orbit insertion ΔV calculations, it is assumed that the full 200 m/sec midcourse propellant is expended. This is a conservative assumption. If midcourse propellant is not expended, orbit insertion ΔV would decrease a small amount, but the orbit trim and vernier capability will increase by a greater amount so that the overall ΔV capability for orbit insertion is increased.

Solid-motor performance is summarized in Table 2.1-4. This motor is a standard Minuteman Wing VI second-stage motor with the following modifications:

Table 2.1-2: 1971 AND 1973 SOLID/LIQUID-SYSTEM PERFORMANCE (3000-POUND CAPSULE)

SYSTEM PERFORMANCE	MIDCOURSE	ORBIT INSERTION: SOLID MOTOR	ORBIT INSERTION: VERNIER	ORBIT INSERTION: SOLID MOTOR + VERNIER	ORBIT TRIM
Maximum Velocity - fps	656	7067	328	7395	328
Minimum Velocity - fps	0.343		0.777	0.777	0.814
Maximum Velocity Tolerance - fps	1.64	40	0.82	0.82	0.82
Acceleration, Maximum - g's	0.0427	3.50	0.097	3.50	0.101
Acceleration, Minimum - g's	0.039		0.092	0.092	0.097

PROPULSION MODULE PERFORMANCE		
Maneuver	Midcourse Correction, Orbit Vernier, and Orbit Trim	Orbit Insertion
Propellant	N_2H_4 Monopropellant	AP - Polybutadiene Binder-AL
Specific Impulse - sec	235	*
Total Impulse Bit - lb/sec	586325	*
Minimum Impulse Bit - lb/sec	200	
Number of Engines	4	1
Thrust per Engine - lb	200	37,570
Engine Operating Time - sec	733	76.5
Thrust Vector Control	Jet Vanes	Freon Fluid Injection
Subsystem Inert Weight - lb	646	1390
Propellant Weight - lb	2495	9839
Subsystem Total Weight - lb	3141	11,229
Structure, Thermal, Power & Contingency - lb	630	
Total Module Weight - lb	15,000	
Total Module Length - in	158	

*See D2-82709-10 Classified Supplement - Reference Page 2

Table 2.1-3: 1971 and 1973 SOLID/LIQUID-SYSTEM PERFORMANCE (2000-POUND CAPSULE)

SYSTEM PERFORMANCE	MIDCOURSE	ORBIT INSERTION			ORBIT TRIM
		SOLID MOTOR	VERNIER	SOLID MOTOR + VERNIER	
Maximum Velocity - fps	656	7615	500	8115	328
Minimum Velocity - fps	0.360		0.87	0.87	0.92
Max Velocity Tolerance - fps	1.64	43	1.25	1.25	0.82
Acceleration, Maximum - g's	0.0449	3.77	0.106	2.77	0.113
Acceleration, Minimum	0.0410		0.103	0.103	0.106

PROPULSION MODULE PERFORMANCE			
Maneuver	Midcourse Correction, Orbit Vernier, and Orbit Trim		Orbit Insertion
Propellant	N_2H_4 Monopropellant		AP - POLYBUTADIENE Binder - AL
Specific Impulse - sec	235		*
Total Impulse - lb/sec	586,325		*
Minimum Impulse Bit - lb/sec	200		
Number of Engines	4		1
Thrust per Engine - lb	200		37,570
Engine Operating Time -sec	733		76.5
Thrust Vector Control	Jet Vanes		Freon Fluid Injection
Subsystem Inert Weight - lb	646		1,390
Propellant Weight - lb	2,495		9,839
Subsystem Total Weight - lb	3,141		11,229
Structure, Thermal, Power & Contingency - lb		630	
Total Module Weight - lb		15,000	
Total Module Length - lb		158	

*See D2-82709-10 Classified Supplement - Reference Page 3

Table 2.1-4: 1971 & 1973 SOLID/LIQUID-SYSTEM SOLID-MOTOR PERFORMANCE (3000-POUND CAPSULE)

Manufacturer	Aerojet
Designation - Original	SR-19-AJ-1
Application - Original	Minuteman Wing VI Second Stage
Average Action Time Thrust - lb	37,570
Total Weight - lb (Incl. TVC)	11,229
Propellant	AP - Polybutadiene Binder - Aluminum
Propellant Weight - lb	9839
Specific Impulse - sec	*
Total Impulse - lb-sec	*
Total Impulse Tolerance - lb-sec	± 17,243
Nozzle Type	Single, Buried Throat, Contoured
Expansion Ratio	32.5
Action Time - sec	76.5
Burning Rate - in/sec	0.337 in/sec @ 500 psia
Chamber Pressure - psia, (Average during action time)	287
Maximum Diameter - in	52
Total Length	150.3
Nozzle Length - in	67.55
λ (Excluding TVC)	0.909
TVC Weight - lb	400

*See D2-82709-10 Classified Supplement - Reference Page 4

1) Case Section

   a) Case shortened 30 inches by removing 3980 lbs of propellant from cylindrical section of grain.

   b) Cork for aerodynamic heating insulation is deleted.

2) Nozzle Section

   a) A 4-pound ring of AISI 416 magnetic steel in the nozzle is replaced with series 300 non-magnetic steel.

   b) A 15-inch contoured extension is added to the existing nozzle increasing the expansion ratio from 24.8 to 32.5.

   c) Cork for aerodynamic heating protection is deleted.

3) TVC

   a) The hot gas roll control system is changed to a regulated cold gas roll control system.

   b) The hot gas TVC pressurization system is changed to stored regulated cold gas.

   c) The hydraulic system pressure supply is changed from a battery-powered electric-motor-driven pump to nitrogen-pressurized freon.

   d) Hydraulic oil for actuation is replaced with filtered Freon 114B2.

   e) The programmed dump system is eliminated.

   f) The steel 17-7PH toroidal freon storage tank is replaced with a titanium storage tank of the same size.

The performance of the hydrazine monopropellant engines is summarized in Table 2.1-5. The design of these engines is based on Ranger and Mariner 50-pound-thrust flight hardware and current 100-pound-thrust engine development work with a spontaneous catalyst.

The weight summaries for the 1971 and 1973 solid/liquid systems for the 2000- and 3000-pound Flight Capsules are shown in Table 2.1-6. System reliability is summarized in Table 2.1-7.

The 1971 and 1973 solid/liquid propulsion system can be used for the 1975 and 1977 missions. Minor changes are required because of increased capsule weight. The modifications are:

1) Solid Motor--Use same case and nozzle; offload 833 pounds of propellant.
2) Liquid Monopropellant System--Add 699 pounds of monopropellant and increase tank size (additional tankage weight is 134 pounds).

Propulsion system performance in 1975 and 1977 is shown in Tables 2.1-8 and 2.1-9. Solid-motor burn time decreases from 76.5 seconds to 71.6 seconds due to offloaded propellant. Monopropellant-engine burn time increases from 733 seconds to 938 seconds.

The 1971-1973 solid/liquid unit, when modified to accommodate a 10,000 lb capsule, has sufficient versatility to attain all feasible orbit sizes and orientations without recourse to orbit insertion vernier. This is because of the low orbit insertion ΔV capability associated with the heavy 1975-1977 planetary vehicle.

Table 2.1-5: 1971 & 1973 SOLID/LIQUID-SYSTEM MONOPROPELLANT-ENGINE PERFORMANCE

Thrust - lb (Vacuum)	200
Propellant	N_2H_4 Monopropellant
Specific Impulse - sec	235
Expansion Ratio	50
Chamber Pressure - psia	150
Engine Weight - lb (incl. TVC Assembly)	20.84
Nozzle Exit Diameter - in.	7.0
Total Length - in.	25.2
Minimum Impulse Bit - lb-sec	50
Minimum Bit Tolerance - lb-sec	$\pm$ 6
Catalyst	Shell 405 (Spontaneous)
Catalyst Weight - lb	3.81
Bed Loading - $Lb/sec/in^2$	0.031
Inlet Valve	Dual - Solenoid Operated
Thrust Vector Control	Jet Vanes - 4 per Engine
TVC Capability - degrees	$\pm$ 5

Table 2.1-6: 1971 & 1973 SOLID/LIQUID-SYSTEM WEIGHT* SUMMARY

Capsule		3,000 & 2,000
Bus and Science		2,500
Propulsion Installation		(15,000)
Midcourse, Orbit Trim and Vernier Propulsion		3,141
Rocket Engine System	87	
Propellant Feed System	17	
Propellant Tanks	210	
Pressurization Feed System	19	
Pressurant and Container	223	
Propellant Residual Allowance	90	
Usable Hydrazine	2,495	
Orbit Insertion Propulsion		11,229
Rocket Motor Inerts	990	
Thrust Vector Control	343	
Roll Control Engine System	5	
TVC and Roll Control N_2 System	52	
Usable Solid Propellant	9,839	
Structure		337
Primary Support Frame	116	
Hydrazine Tank Support	12	
Nitrogen Tank Support	3	
Solid Motor Support	64	
Midcourse Engine Thrust Structure	27	
Meteoroid Shielding, Bus-Capsule	32	
Meteoroid Shielding, Thermal Shield	34	
Meteoroid/Thermal Support Structure	38	
Miscellaneous Support Structure	11	
Thermal Control		118
Shell	46.6	
Capsule-Bus	14.4	
Propulsion	8.7	
Solar Shield	19	
Thermal Shield	26	
Instrumentation, Heaters and Switches	3.3	
Cabling and Power Conditioning		25
Converters and Switch Installation	11.4	
Cable Harness	13.6	
Contingency**		150
Planetary Vehicle Gross Weight		20,500 & 19,500

*All weights are expressed in pounds
**Contingency includes a 3 percent allowance for weight growth of developed hardware and a 10 percent allowance for new hardware.

Table 2.1-7: 1971 & 1973 SOLID/LIQUID-SYSTEM RELIABILITY SUMMARY

ITEM	$\lambda \times 10^6$	N	t	Failures/10^6 λN t$\times 10^6$
Monopropellant System				
Piping, Tanks & Connections -N_2	0.167/hr	1	5112 hrs	855
Valve-Squib-N.O.-N_2	25/cy	Redun	1 cy	Negl.
Valve-Squib-N.O.-N_2	25/cy	Redun	1 cy	Negl.
Valve-Squib-N.C.-N_2	25/cy	Redun	1 cy	Negl.
Valve-Solenoid Latch-N_2				
- Closed	1/cy	Redun	3 cy	Negl.
- Open	1/cy	2	3 cy	6
- Leak, Backed by regulator	.5/hr	Redun	4364 hrs	Negl.
Filter-N_2	3.3/hr	1	0.3 hr	1
Regulator-Pressure-N_2	2.4/hr	1 Quad	0.3 hr	Negl.
Tank N_2H_4	0.0047/hr	4	5112 hrs	96
Bladder-N_2H_4 Expulsion	200/cy	4	1 cy	800
Filter-N_2H_4	3.3/hr	2	0.3 hr	2
Valve-Squib-N.C.-N_2H_4	25/cy	Redun	1 cy	Negl.
Valve-Squib-N.O.-N_2H_4	25/cy	Redun	1 cy	Negl.
Valve-Squib-N.C.-N_2H_4	25/cy	Redun	1 cy	Negl.
Valve-Solenoid Latch-N_2H_4				
- Closed	1/cy	Redun	4 cy	Negl.
- Open	1/cy	Redun	4 cy	Negl.
- Leak	5/hr	Redun	4364 hrs	Negl.
Check Valve-Thermal Relief	1/hr	1 Quad	0.3 hrs	Negl.
Relief Valve & Burst Disk	0.001/hr	2	5112 hrs	10
Orifice	0.15/hr	4	0.3 hr	Negl.
Engine-N_2H_4	100/cy	3 of 4	4 cy	Negl.
Jet Vane Assembly-TVC	15/hr	3 of 4	0.3 hrs	Negl.
			TOTAL	1770
			R_e =	0.9982
Solid System				
Motor-Solid Rocket	50/cy	1	1 cy	50
Piping, Tanks & Connections-N_2				
- TVC	0.17/hr	2	5040 hrs	847
Valve-Squib-N.C.-N_2	25/cy	Redun	1 cy	Negl.
Regulator-Pressure-N_2	2.4/hr	1 Quad	0.1 hr	Negl.
Tank-Freon w/Bladder	200/cy	1	1 cy	200
Torque Motor	4.5/cy	4	10 cy	180
Valve-Injector-Freon	25/cy	4	10 cy	1000
Valve-Solenoid-Roll	2/cy	2 Quad	10 cy	Negl.
Thruster-Roll Control	0.4/hr	2	0.1 hr	Negl.
			TOTAL	2277
			R_e =	0.9977
	SOLID/LIQUID SYSTEM R_e =			0.9960

λ = Failures/Hr or Cycle
t = Hours or Cycles
N = Number of Components

Table 2.1-8: 1971 & 1973 SOLID/LIQUID-SYSTEM PERFORMANCE--SOLID OFFLOADED FOR 1975 TO 1977 MISSION (10,000-POUND CAPSULE)

SYSTEM PERFORMANCE	MIDCOURSE	ORBIT INSERTION	ORBIT TRIM
Maximum Velocity - fps	656	4012	328
Minimum Velocity - fps	0.250		0.401
Maximum Velocity Tolerance-fps	1.64	31	0.82
Acceleration, Maximum - g's	0.031	2.38	0.0498
Acceleration, Minimum - g's	0.028		0.048

PROPULSION MODULE PERFORMANCE		
Maneuver	Midcourse Correction and Orbit Trim	Orbit Insertion
Propellant	N_2H_4 Monopropellant	AP - Polybutadiene Binder - Al
Specific Impulse - sec	235	*
Total Impulse - lb-sec	750,590	*
Minimum Impulse Bit - lb-sec	200	
Number of Engines	4	1
Thrust per Engine - lb	200	36,741
Engine Operating Time - sec	938	71.6
Thrust Vector Control	Jet Vanes	Fluid Injection
Subsystem Inert Weight - lb	772	1390
Propellant Weight - lb	3194	9006
Subsystem Weight - lb	3966	10,396
Structure, Thermal, Power and Contingency	638	
Total Module Weight - lb	15,000	
Total Module Length - in	150	

*See D2-82709-10 Classified Supplement - Reference Page 5

Table 2.1-9: 1971 & 1973 SOLID/LIQUID-SYSTEM PERFORMANCE--SOLID OFFLOADED FOR 1975 to 1977 MISSION (8000-POUND CAPSULE)

		ORBIT INSERTION			
SYSTEM PERFORMANCE	MIDCOURSE	SOLID MOTOR	VERNIER	SOLID MOTOR + VERNIER	ORBIT TRIM
Maximum Velocity - fps	656	4394	142	4536	328
Minimum Velocity - fps	0.266		.43	.43	.46
Maximum Velocity Tolerance - fps	1.64	32	0.35	11.3	0.82
Acceleration, Maximum - g's	0.033	2.62	.0542	2.62	.0571
Acceleration, Minimum - g's	0.0302		.0534	.0534	.0542

PROPULSION MODULE PERFORMANCE		
Maneuver	Midcourse Correction and Orbit Trim	Orbit Insertion
Propellant	N_2H_4 Monopropellant	AP-Polybutadiene Binder-Al
Specific Impulse, sec	235	*
Total Impulse, lb/sec	750,590	*
Minimum Impulse Bit, lb/sec	200	
Number of Engines	4	1
Thrust per Engine, lb	200	36,741
Engine Operating Time, sec	938	71.6
Thrust Vector Control	Jet Vanes	Fluid Injection
Subsystem Inert Weight, lb	772	1390
Propellant Weight, lb	3194	9006
Subsystem Weight, lb	3966	10,396
Structure, Thermal, Power & Weight Contingency	638	
Total Module Weight, lb	15,000	
Module Length, in	158	

*See D2-82709-10 Classified Supplement - Reference Page 6

2.2 OPTIMUM SOLID/LIQUID SYSTEM SIZED FOR 1975 AND 1977 MISSIONS

The optimum 1975 and 1977 solid/liquid system is similar to the optimum 1971 and 1973 solid/liquid system. Major differences required to accommodate increased capsule weights are:

1) Increased amount of hydrazine for midcourse correction and orbit trim maneuvers;
2) Smaller orbit insertion system due to the allocated 15,000-pound propulsion limit;
3) Structural changes resulting from added propellant and capsule weight.

The 1975 and 1977 solid/liquid configuration, shown in Figure 2.2-1, is similar to the 1971 and 1973 configuration. The only noticeable external differences are the size of the monopropellant tanks (increased for the larger amount of hydrazine required) and a 6" shorter motor case. The same four 200-pound-thrust, radiation-cooled, monopropellant engines and jet vane thrust vector control assemblies are installed in this configuration.

The orbit-insertion freon liquid injection thrust vector control system is identical to that used in 1971 and 1973. The schematic and parts list for the 1975 and 1977 system are identical to that shown for the 1971 and 1973 system in Figure 2.1-2 and Table 2.1-1.

The 1975 and 1977 solid/liquid system performance summaries for the 10,000- and 8000-pound capsules are given in Tables 2.2-1 and 2.2-2, respectively.

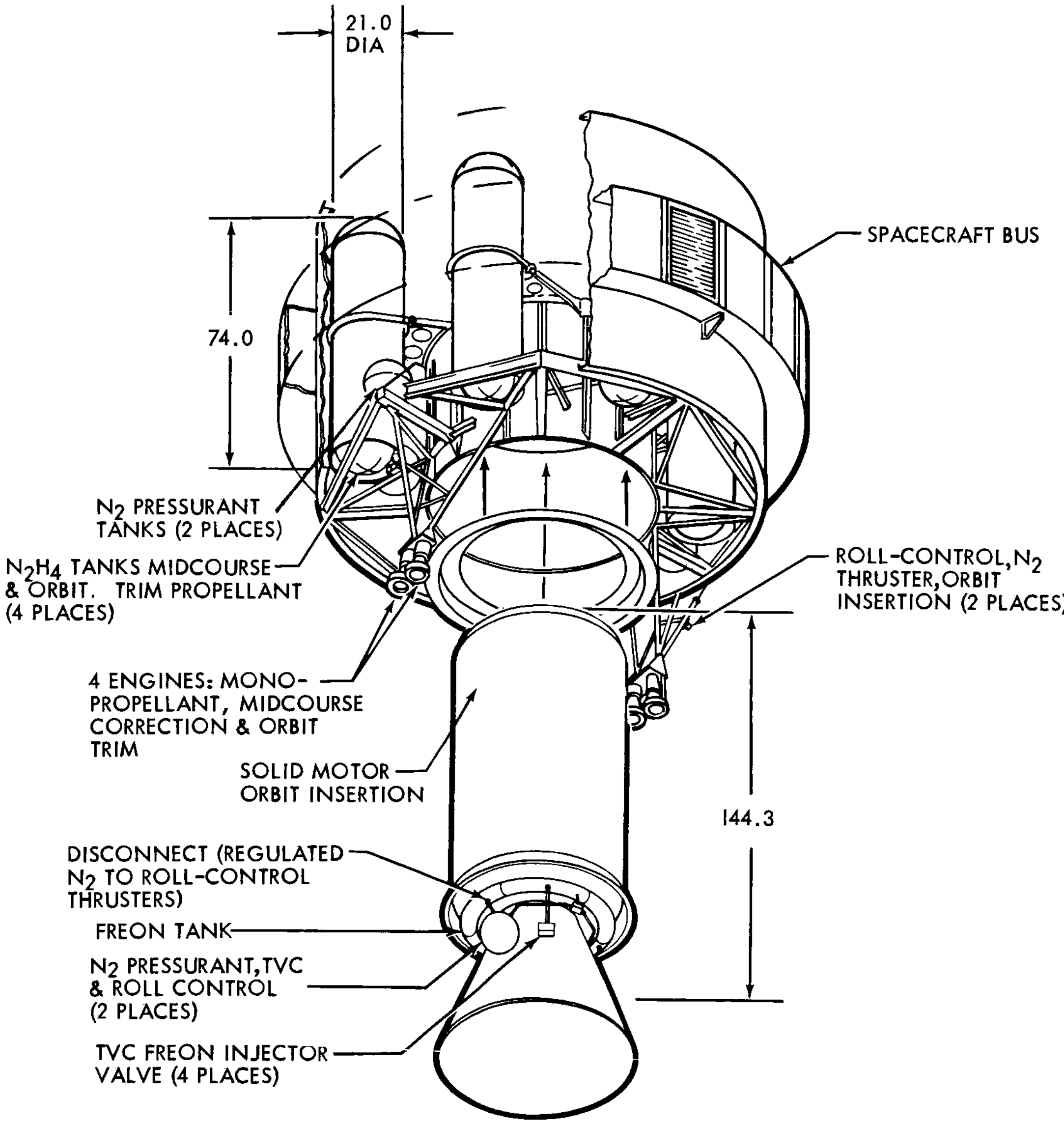


Figure 2.2-1: Solid/Liquid Propulsion System
1975 - 1977 Module

Table 2.2-1: 1975 & 1977 SOLID/LIQUID-SYSTEM PERFORMANCE (10,000-POUND CAPSULE)

SYSTEM PERFORMANCE	MIDCOURSE	ORBIT INSERTION	ORBIT TRIM
Maximum Velocity - fps	656	4046	328
Minimum Velocity - fps	0.25		0.403
Maximum Velocity Tolerance-fps	1.64	30	0.82
Acceleration, Maximum - g's	0.031	2.24	0.050
Acceleration, Minimum - g's	0.028		0.048

PROPULSION MODULE PERFORMANCE		
Maneuver	Midcourse Correction and Orbit Trim	Orbit Insertion
Propellant	N_2H_4 Monopropellant	AP - Polybutadiene Binder - Al
Specific Impulse - sec	235	*
Total Impulse - lb-sec	749,650	*
Minimum Impulse Bit - lb-sec	200	
Number of Engines	4	1
Thrust per Engine - lb	200	33,786
Engine Operating Time - sec	937	78.2
Thrust Vector Control	Jet Vanes	Fluid Injection
Subsystem Inert Weight - lb	772	1355
Propellant Weight - lb	3190	9045
Subsystem Total Weight - lb	3962	10,400
Structure, Thermal, Power & Contingency	638	
Total Module Weight - lb	15,000	
Total Module Length - in	158	

*See D2-82709-10 Classified Supplement - Reference Page 7

Table 2.2-2: 1975 & 1977 SOLID/LIQUID-SYSTEM PERFORMANCE (8000-POUND CAPSULE)

SYSTEM PERFORMANCE	MIDCOURSE	ORBIT INSERTION: SOLID MOTOR	ORBIT INSERTION: VERNIER	ORBIT INSERTION: SOLID MOTOR + VERNIER	ORBIT TRIM
Maximum Velocity - fps	656	4434	133	4567	328
Minimum Velocity - fps	.266		.440	.440	.460
Maximum Velocity Tolerance-fps	1.64	32	0.33	0.33	0.82
Acceleration, Maximum - g's	.033	2.44	.0545	2.44	.0572
Acceleration, Minimum - g's	.0302		.0535	.0535	.0545

PROPULSION MODULE PERFORMANCE			
Maneuver	Midcourse Correction, Orbit Orbit Vernier and Orbit Trim		Orbit Insertion
Propellant	N_2H_4 Monopropellant		AP - Polybutadiene Binder - Al
Specific Impulse - sec	235		*
Total Impulse - lb-sec	749,650		*
Minimum Impulse Bit - lb-sec	200		
Number of Engines	4		1
Thrust per Engine - lb	200		33,786
Engine Operating Time - sec	937		78.2
Thrust Vector Control	Jet Vanes		Fluid Injection
Subsystem Inert Weight - lb	772		1355
Propellant Weight - lb	3190		9045
Subsystem Total Weight - lb	3962		10,400
Structure, Thermal, Power & Contingency-lb		638	
Total Module Weight - lb		15,000	
Total Module Length - in		158	

*See D2-82709-10 Classified Supplement - Reference Page 8

The 1975 and 1977 solid/motor performance is shown in Table 2.2-3. Beryllium, instead of aluminum as a propellant metal additive, was considered for the 1975 and 1977 missions. It was found that the thermal radiation from a beryllium exhaust plume increased solar panel heating beyond currently acceptable limits. Beryllium was therefore rejected.

Table 2.2-3: 1975 & 1977 SOLID/LIQUID SYSTEM--SOLID-MOTOR PERFORMANCE (10,000-POUND CAPSULE)

Manufacturer	Aerojet
Designation - Original	SR-19-AJ-1
Application - Original	Minuteman Wing VI Second Stage
Average Action Time Thrust - lb(Vacuum)	33,786
Total Weight - lb (Inc. TVC)	10,400
Propellant	AP - Polybutadiene Binder - Aluminum
Propellant Weight - lb	9045
Specific Impulse - sec	*
Total Impulse - lb-sec	*
Total Impulse Tolerance - lb-sec	± 15,852
Nozzle Type	Single - Buried Throat, Contoured
Expansion Ratio	32.5
Action Time - sec	78.2
Burning Rate - in/sec	0.337 in/sec @ 500 psia
Average Action Time Chamber Pressure - psia	255
Maximum Diameter - in.	52
Total Length - in.	144.3
Nozzle Length - in.	67.55
λ (Excluding TVC)	0.904
TVC Weight - lb	390

*See D2-82709-10 Classified Supplement - Reference Page 9

The 1975 & 1977 monopropellant engine performance is identical to that shown for the 1971 & 1973 engines in Table 2.1-5. The longer engine burn time required in the 1975 & 1977 missions is well within engine capability.

Weight summaries for the 1975 & 1977 solid/liquid system for 10,000- and 8,000-pound capsules are shown in Table 2.2-4.

System reliability is comparable to that of the 1971 & 1973 solid/liquid system. The components are not changed, and the mission profile is similar.

Use of the 1975 & 1977 solid/liquid propulsion system for the 1971 & 1973 missions requires no propulsion system changes. Excess monopropellant system capability is used to augment total vehicle ΔV for orbit insertion, i.e. orbit insertion vernier. System performance in 1971 & 1973 is shown in Tables 2.2-5 and 2.2-6. Since the solid motor is not changed, the motor burn time remains constant. No resizing is required for the 1971 and 1973 missions.

Table 2.2-4: 1975 & 1977 SOLID/LIQUID-SYSTEM WEIGHT* SUMMARY

Capsule		10,000 & 8,000
Bus and Science		3,500
Propulsion Installation		(15,000)
Midcourse and Orbit Trim Propulsion		3,962
Rocket Engine System	87	
Propellant Feed System	17	
Propellant Tanks	263	
Pressurization Feed System	19	
Pressurant and Container	282	
Propellant Residual Allowance	104	
Usable Hydrazine	3190	
Orbit Insertion Propulsion		10,400
Rocket Motor Inerts	965	
Thrust Vector Control	333	
Roll Control Engine System	5	
TVC and Roll Control N_2 System	52	
Usable Solid Propellant	9045	
Structure		337
Primary Support Frame	116	
Hydrazine Tank Support	12	
Nitrogen Tank Support	3	
Solid Motor Support	64	
Midcourse Engine Thrust Structure	27	
Meteoroid Shielding, Bus-Capsule	32	
Meteoroid Shielding, Thermal Shield	34	
Meteoroid/Thermal Support Structure	38	
Miscellaneous Support Structure	11	
Thermal Control		118
Shell	46.6	
Capsule-Bus	14.4	
Propulsion	8.7	
Solar Shield	19	
Thermal Shield	26	
Instrumentation, Heaters and Switches	3.3	
Cabling and Power Conditioning		25
Converters and Switch Installation	11.4	
Cable Harness	13.6	
Contingency**		158
Planetary Vehicle Gross Weight		28,500 & 26,500

*All weights expressed in pounds
**Contingency includes a 3 percent allowance for weight growth of developed hardware and a 10 percent allowance for new hardware.

Table 2.2-5: 1975 & 1977 SOLID/LIQUID-SYSTEM USED FOR 1971 & 1973 MISSION PERFORMANCE (3000-POUND CAPSULE)

SYSTEM PERFORMANCE	MIDCOURSE	ORBIT INSERTION: SOLID MOTOR	ORBIT INSERTION: VERNIER	ORBIT INSERTION: SOLID MOTOR + VERNIER	ORBIT TRIM
Maximum Velocity - fps	656	6260	877	7137	328
Minimum Velocity - fps	0.343		.752	.752	.792
Maximum Velocity Tolerance - fps	1.64	40	2.19	2.19	.82
Acceleration, Maximum - g's	0.0427	3.30	.0935	.0935	.0985
Acceleration, Minimum - g's	0.039		.0835	.0835	.0935

PROPULSION MODULE PERFORMANCE			
Maneuver	Midcourse Correction, Orbit Vernier and Orbit Trim		Orbit Insertion
Propellant	N_2H_4 Monopropellant		AP- Polybutadiene Binder - Al
Specific Impulse - sec	235		*
Total Impulse - lb-sec	749,650		*
Minimum Impulse Bit - lb-sec	200		
Number of Engines	4		1
Thrust per Engine - lb	200		33,786
Engine Operating Time - sec	937		78.2
Thrust Vector Control	Jet Vanes		Fluid Injection
Subsystem Inert Weight - lb	772		1355
Propellant Weight - lb	3190		9045
Subsystem Weight - lb	3962		10,400
Structure, Thermal, Power & Contingency		638	
Total Module Weight - lb		15,000	
Module Length - in		158	

*See D2-82709-10 Classified Supplement - Reference Page 10

Table 2.2-6: 1975 & 1977 SOLID/LIQUID SYSTEM USED FOR 1971 & 1973 MISSIONS PERFORMANCE (2000-POUND CAPSULE)

SYSTEM PERFORMANCE	Midcourse	Orbit Insertion: Solid Motor	Orbit Insertion: Vernier	Orbit Insertion: Solid Motor +Vernier	Orbit Trim
Maximum Velocity -fps	656	6725	1108	7833	328
Minimum Velocity -fps	0.361		0.875	0.875	0.891
Maximum Velocity Tolerance-fps	1.64	43	0.277	0.277	0.82
Acceleration, Maximum - g's	0.0449	3.50	0.109	0.109	0.114
Acceleration, Minimum - g's	0.0410		0.0935	0.0935	0.109

PROPULSION MODULE PERFORMANCE		
Maneuver	Midcourse Correction, Orbit Vernier and Orbit Trim	Orbit Insertion
Propellant	N_2H_4 Monopropellant	AP - Polybutadiene Binder -AL
Specific Impulse- sec	235	*
Total Impulse - lb sec	749650	*
Minimum Impulse Bit - lb sec	200	
Number of Engines	4	1
Thrust per Engine - lb	200	33,786
Engine Operating Time - sec	937	78.2
Thrust Vector Control	Jet Vanes	Fluid Injection
Subsystem Inert Weight - lb	772	1355
Propellant Weight - lb	3190	9045
Subsystem Weight - lb	3962	10,400
Structure, Thermal, Power & Contingency	638	
Total Module Weight - lb	15,000	
Module Length - in.	158	

*See D2-82709-10 Classified Supplement - Reference Page 11

2.3 OPTIMUM LEM DESCENT PROPULSION SYSTEM

The lunar excursion module (LEM) descent stage provides the propulsion system and structural support for lunar landing. The basic shape is an octagonal box with flat ends. Primary structure is aluminum alloy beams and panels. The entire structural box is covered with 2 inches of insulation and a very thin aluminum alloy outer skin for thermal protection. Overall dimensions are 166 inches across the octagonal flats and 104 inches deep. For the LEM application, the descent stage contains the descent engine and its associated propulsion subsystems. Control instrumentation, scientific equipment, and storage tanks for water, oxygen, and hydrogen used by the LEM environmental control and electrical power subsystems are also included. When adapted to the Voyager mission, as shown in Figure 2.3-1, it contains only the main propulsion subsystem and a supplementary thrust vector control subsystem, which is also used for propellant settling.

The LEM descent engine is mounted in the center compartment of the structure. It is surrounded by four propellant tanks, two for fuel and two for the oxidizer. The propellant tanks contain slosh and anti-vortex baffles and are interconnected at the top and bottom for liquid level equalization. The propellant tanks are pressurized by regulated helium contained in two ambient-temperature high-pressure storage vessels. A schematic diagram of the propulsion system is shown in Figure 2.3-2, and the components are listed in Table 2.3-1. The required propellant positioning and thrust vector control are accomplished by a

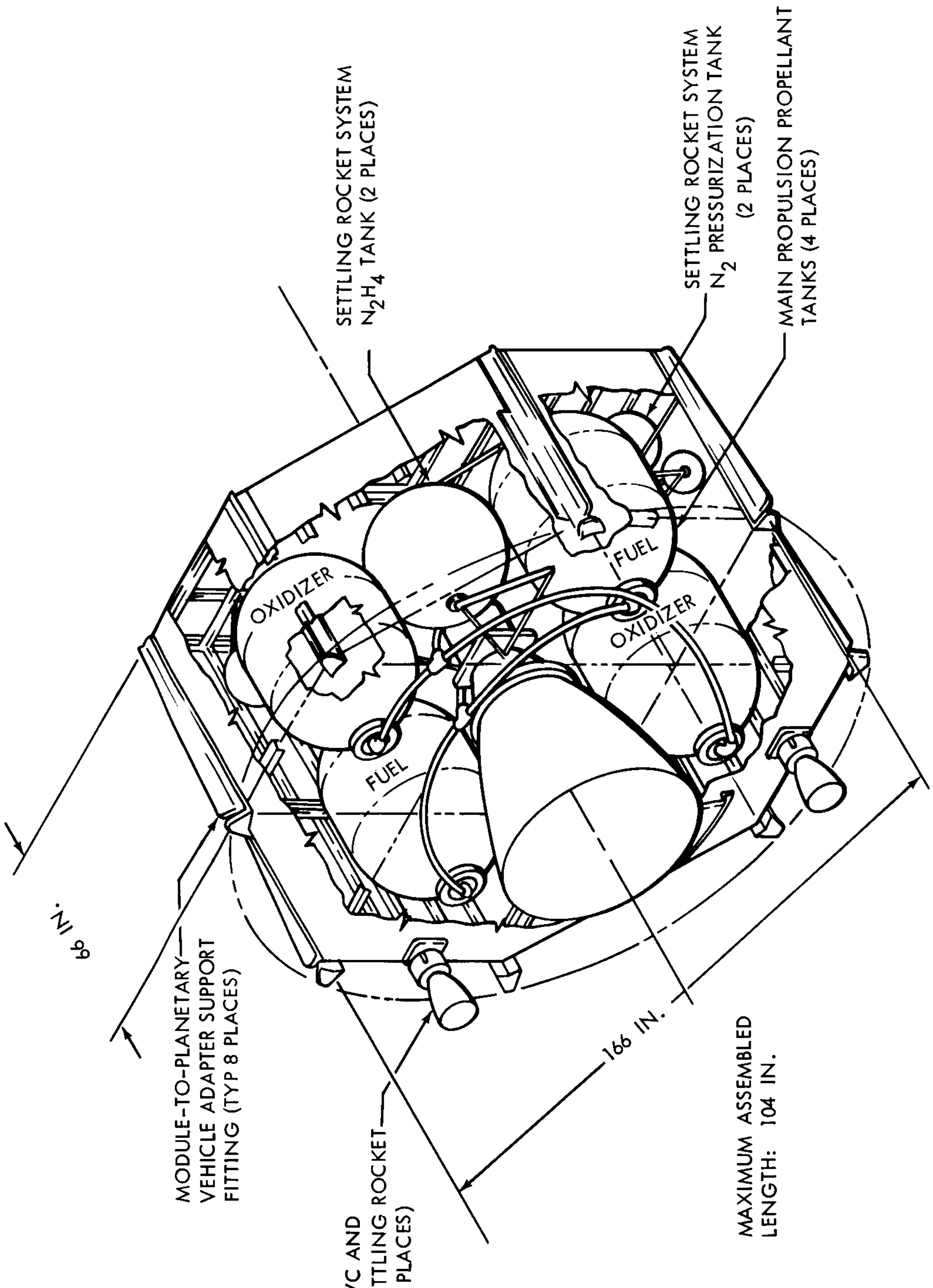


Figure 2.3-1: LEM Propulsion Module For Voyager Application

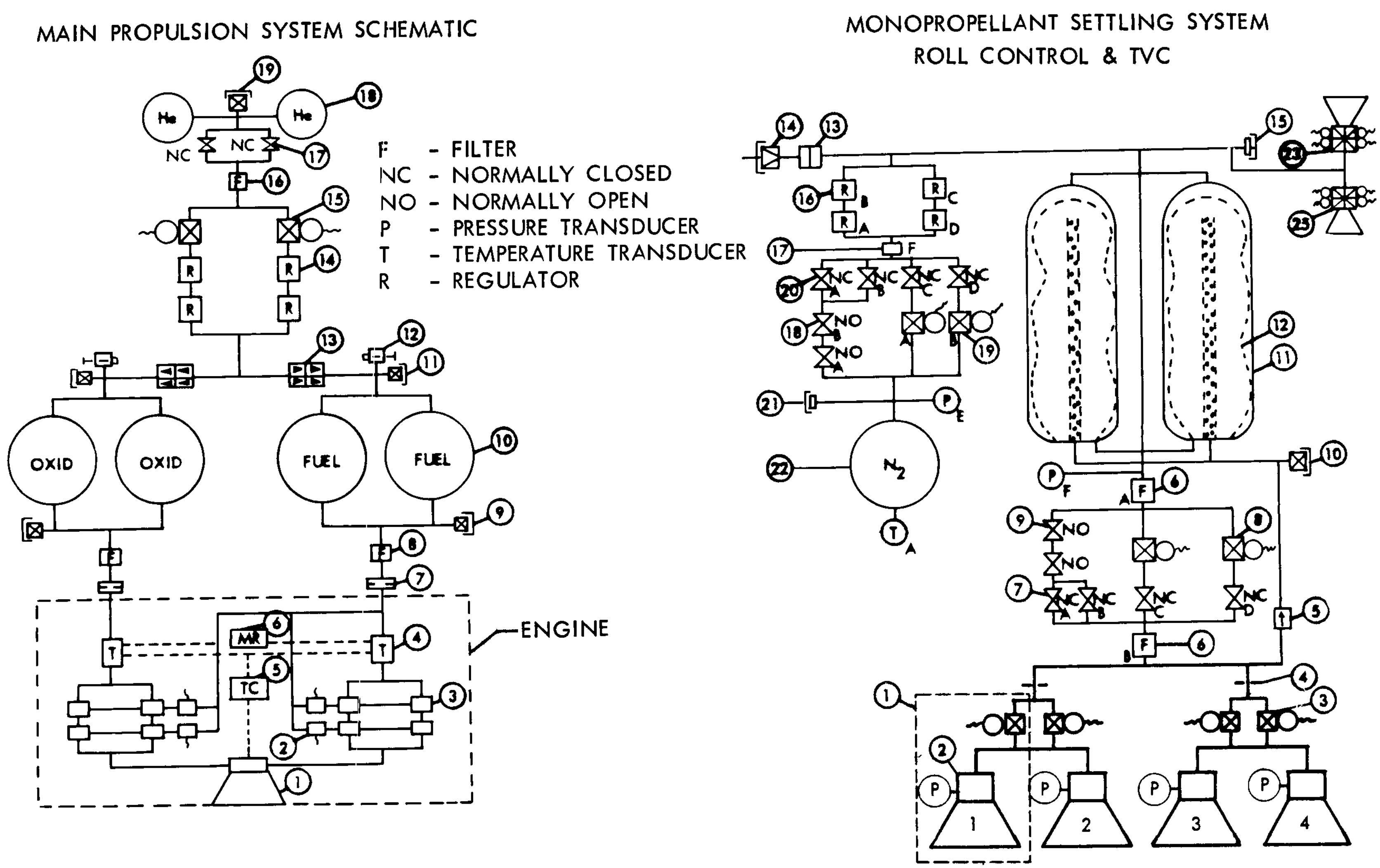


Figure 2.3-2: Modified LEM Propulsion System

Table 2.3-1: MODIFIED LEM PROPULSION SYSTEM--COMPONENT LIST

MAIN PROPULSION SYSTEM

ITEM	QTY.	NAME
1	1	Engine Nozzle and Thrust Chamber
2	4	Engine Pilot Valves and Solenoids
3	8	Propellant Shut-off Valves
4	2	Throttle Valves
5	1	Throttle Control Actuator
6	1	Mixture Ratio Control Actuator
7	2	Trim Orifice
8	2	Propellant Filter
9	2	Valve and Cap, Fill and Drain
10	4	Tank, propellant
11	2	Vent Valve
12	2	Relief Valve and Burst Disk
13	2	Quad Check Valve (Assembly)
14	4	Regulator, Pressure
15	2	Solenoid Valve, Latching
16	1	Filter, Helium
17	2	Explosive Valve, Dual Squib,N.C.
18	2	Tank, Pressurization Gas
19	1	Valve and Cap, Fill and Drain, Helium
20		Structural Skin
21		Insulation and Thin Skin
22		Main Structural Support
23		Thermal Control
24		Meteoroid Shield

Table 2.3-1 (CONT.): MODIFIED LEM PROPULSION SYSTEM--COMPONENT LIST

TVC AND MONOPROPELLANT SETTLING PROPULSION SYSTEM

ITEM	QTY.	NAME
1	4	Rocket Engine Assembly
2	4	Catalyst Bed
3	4	Valve, Solenoid
4	4	Orifice
5	1	Thermal Relief
6	2	Filter, Propellant
7	4	Valve, Squib, N.C.
8	2	Valve, Latching, Solenoid
9	2	Valve, Squib, N.O.
10	1	Valve and Cap, Fill and Drain
11	2	Tank, Propellant
12	2	Bladder, Positive Expulsion
13	1	Burst Disk
14	1	Valve, Relief
15	1	Valve and Cap, Vent and Press.
16	4	Regulator Nitrogen
17	1	Filter, Nitrogen
18	2	Valve, Squib, N.O.
19	2	Valve, Latching Solenoid
20	4	Valve, Squib, N.C.
21	1	Valve, and Cap, Press.
22	1	Tank, Nitrogen
23	2	Roll Control Thrusters and Quad Valves

separate pulsing monopropellant propulsion subsystem consisting of four 100-pound-thrust hydrazine engines. The necessity for using this pulsing system for thrust vector control is explained in Section 6.3.

Voyager spacecraft performance using the LEM descent propulsion system as the propulsion module is tabulated in Table 2.3-2 for the 3000-pound capsule and Table 2.3-3 for the 2000-pound capsule. Main propulsion engine and settling engine performance characteristics are included in Tables 2.3-4 and 2.3-5. The propulsion module weight statement is shown in Table 2.3-6. A summary of the LEM descent system reliability study is shown in Table 2.3-7.

For the 1975 and 1977 missions the LEM descent propulsion system provides settling and thrust vector control for midcourse, orbit insertion and orbit trim for a heavier planetary vehicle. With an 8000 to 10,000 pound capsule, the amount of hydrazine is increased from 144 to 166 pounds. To stay within the 15,000 pound allocation, bipropellant weight is decreased. System performance in 1975 and 1977 is shown in Tables 2.3-8 and 2.3-9.

A weight statement for the 1975 and 1977 missions is shown in Table 2.3-10.

The reliability summary for the 1971 and 1973 LEM system also applies for system use in 1975 and 1977 because system components and operating conditions are similar.

Table 2.3-2: 1971 & 1973 MODIFIED LEM DESCENT PROPULSION MODULE SYSTEM PERFORMANCE (3,000-POUND CAPSULE)

System Performance	Midcourse	Orbit Insertion	Orbit Trim
Maximum Velocity, fps	656	6935	328
(1) Minimum Velocity, fps	4.17	Not Applicable	9.16
(2) Minimum Velocity, fps	0.015	Not Applicable	0.033
Maximum Velocity Tolerance, fps	0.168	0.346	0.357
Acceleration, Maximum, g's (Max. thrust)	0.548	1.13	1.16
Acceleration, Minimum, g's (Min. thrust)	0.0512	0.0558	0.113

Propulsion Module Performance		
Maneuver	Propellant Settling & Thrust Vector Control	Midcourse, Orbit Insertion and Orbit Trim
Propellant	N_2H_4	N_2O_4/Aerozine 50
Specific Impulse, sec	(3) 230 avg.	*
Total Umpulse, lb-sec	33,100	*
Minimum Impulse Bit, lb-sec	2.0 ± .2	248 (est.)
Start Impulse Tolerance, lb-sec	0.1	±100
Shutdown Impulse Tolerance, lb-sec	0.1	±100
Number of Engines	4	1
Thrust per Engine, lb	100	10,500 - 1050
Engine Operating Time, sec (Each)	85	320 (Max. Thrust)
Thrust Vector Control	Pulse Mode	Pulse Monopropellant System
Subsystem Inert Weight, lb	240	3244
Propellant Weight, lb	144	11,372
Subsystem Weight, lb	384	14,616
Total Module Weight, lb	15,000	
Module Length, in.	150	

(1) With Minimum-Thrust Main Engine and Propellant Settling Engines
(2) With Propellant Settling Engines Only
(3) Corrected for Pulse Modes

*See D2-82709-10 Classified Supplement - Reference Page 12

TABLE 2.3-3: 1971 & 1973 LEM DESCENT PROPULSION MODULE SYSTEM PERFORMANCE (2000-POUND CAPSULE)

System Performance	Miscourse	Orbit Insertion	Orbit Trim
Maximum Velocity, fps	656	7597	328
(1) Minimum Velocity, fps	4.38	Not Applicable	10.3
(2) Minimum Velocity, fps	0.016	Not Applicable	0.037
Maximum Velocity Tolerance, fps	0.177	0.389	0.401
Acceleration, Maximum, g's (Max. thrust)	0.575	1.27	1.31
Acceleration, Minimum, g's (Min. Thrust)	0.0539	0.0588	0.127

Propulsion Module Performance		
Maneuver	Propellant Settling and Thrust Vector Control	Midcourse, Orbit Insertion and Orbit Trim
Propellant	$N_2 H_4$	N_2O_4/Aerozine 50
Specific Impulse, sec	(3) 230 avg.	*
Total Impulse, lb-sec	33,100	*
Minimum Impulse Bit, lb-sec	2.0 ± .2	248 (est.)
Start Impulse Tolerance, lb-sec	0.10	±100
Shutdown Impulse Tolerance, lb-sec	0.10	±100
Number of Engines	4	1
Thrust per Engine, lb	100	10,500 - 1050
Engine Operating Time, sec (each)	85	320 (max. thrust)
Thrust Vector Control	Pulse Mode	Pulse Monopropellant System
Propellant Weight, lb	144	11,372
Subsystem Inert Weight, lb	240	3244
Subsystem Weight, lb	384	14,616
Total Module Weight, lb	15,000	
Module Length, in	150	

(1) With Minimum Thrust Main Engine and Propellant Settling Engines
(2) With Propellant Settling Engines Only
(3) Corrected For Pulse Modes

*See D2-82709-10 Classified Supplement - Reference Page 13

Table 2.3-4: MODIFIED LEM DESCENT PROPULSION MODULE

MAIN ENGINE DATA SHEET

Designation	LEMDE (LEM Descent)
Manufacturer	STL
Status	Development
Propellants — Fuel	Aerozine-50
Propellants — Oxidizer	N_2O_4
Engine Thrust	10,500 - 1,050 (Vacuum)
Engine Specific Impulse	*
Mixture Ratio O/F	1.6 ± 0.02 at F_{max} and ± 0.06 at F_{min}
Expansion Ratio A_e/A_t	47.5
Exit Area	2583 In^2
Chamber Pressure	110 to 11 psia
Start time and Impulse	*
Shutdown Time and Impulse	*
Minimum Total Impulse Bit	*
Throttle Ratio	10/1 continuous
Restart Capability	Multiple (20)
Burn Time and Service Life	1200 Sec, 20 starts
Ignition	Hypergolic
Cooling	Ablative/Radiation
Weight, Dry	398.7 Lbs.
Size — Length	85.0 Inches
Size — Diameter	58.26 Inches
Thrust Vector — Type	Pulse Mode Operation of Propellant Settling and Thrust Vector Control (TVC) Engines
Fuel Inlet Pressure	220 psia
Oxidizer Inlet Pressure	220 psia

*See D2-82709-10 Classified Supplement - Reference Page 14

Table 2.3-5: MODIFIED LEM DESCENT PROPULSION MODULE THRUST VECTOR CONTROL AND PROPELLANT SETTLING ENGINE DATA SHEET

Designation		
Manufacturer		
Status	New development required	
Propellant	N_2H_4 Monopropellant	
Engine Thrust	100 LBF	
Engine Specific Impulse	235 Seconds	(Steady State)
Mixture Ratio O/F	Monopropellant	
Expansion Ratio A_e/A_t	50	
Exit Area	28.4 sq in	
Chamber Pressure	150 psia	
Start Time and Impulse (S.S.)	22 ms On to First Rise (Cold) 300 ms First Rise to 9 percent (Cold)	
Shutdown Time and Impulse (S.S.)	10 ms Off to First Drop 125 ms First Drop to 10 percent	
Minimum Total Impulse Bit	2 $\pm$.2 LB-SEC	(Excluding Pulse Mode)
Throttle Ratio	None	
Restart Capability	Multiple	
Burn Time or Service Life	500 to 1000 sec	
Ignition	Spontaneous Catalyst	
Cooling	Radiation	
Weight, Dry	5.3 lb	
Size Length Diameter	8.2 in. 6.1 in.	
Thrust Vector Type	Pulse Mode Operation	
Fuel Inlet Pressure	260 psi Nominal	

Table 2.3-6: 1971 TO 1973 LEM PROPULSION SYSTEM WEIGHT+ SUMMARY

Capsule	2000 and 3000
Allocated Bus and Science	2500
Propulsion Installation	(15000)
Engine System	405
*Propellant Supply System	510
*Pressurization System	530
Structure	1120
*Adapter Fittings	120
*Trapped Propellant	165
*Roll Control System	31
*Monopropellant System Inerts	240
*Monopropellant	144
*Instrumentation Sensors & Wiring	92
Bipropellant Leakage	282
Usable Bipropellant	11090
*Additional Thermal Protection	105
**Contingency	166
Planetary Vehicle Gross Weight	19,500 and 20,500

+All Weights in Pounds
*LEM Modifications
**Contingency Includes a 3 percent Allowance for Weight Growth of Developed Hardware and a 10 percent Allowance for New Hardware.

Table 2.3-7: 1971 & 1973 CONFIGURATION LEM DESCENT PROPULSION SYSTEM RELIABILITY SUMMARY

Item	$\lambda \times 10^6$	N	t	Failures/10^6 $\lambda N t \times 10^6$
LEM Engine				
Piping, Tanks & Connections	0.167/hr	1	5112 hrs	855
Fill Valves (Capped)	Negl.			Negl.
Dual Squib Expulsion Valve	28/cy	1	1 cy	28
Filter, N_2	3.3/hr	1	0.9 hr	3
Solenoid Valves	1/cy	Redun legs		Negl.
Pressure Regulator	2.4/hr		0.1 hr	23
Check Valves	1/hr	2 Quad	0.1 hr	1
Burst Disk and Rel. Valve	0.0017/hr	2	5112 hrs	17
Propellant Tanks	0.0835/hr	2	5112 hrs	1710
Fill and Vent (capped)	Negl.			Negl.
Fuel Filters and Orfices	3.3/hr	2	0.9 hr	6
Bipropellant Engine (including valves and throttling)	270/cy	1	5 cy	1350
			TOTAL	3993
			$R_e = 0.996$	
Settling and TVC				
Monopropellant System	0.268/hr	1 syst	5112 Hrs	1370
Engines for Settling (starts)	100/cy	4	5 cy	2000
Engines in Pulse Mode (for Thrust Vector Control)	6650/hr		0.086 hr	2283
Roll Control Valves	2/cy	Redun	10 cy	Negl.
Roll Control Thrusters	0.4/hr	2	0.1 hr	Negl.
			TOTAL	5653
			$R_e = 0.994$	
LEM plus Monopropellant System			$R_e = 0.9904$	

λ = Failures/Hr or Cycle
N = Number of Components
t = Hours or Cycles

Table 2.3-8: 1975 & 1977 MODIFIED LEM DESCENT PROPULSION MODULE SYSTEM PERFORMANCE (10,000-POUND CAPSULE)

System Performance	Midcourse	Orbit Insertion	Orbit Trim
Maximum Velocity, fps	656	3972	328
(1)Minimum Velocity, fps	3.00	Not Applicable	4.86
(2)Minimum Velocity, fps	0.011	Not Applicable	0.018
Maximum Velocity Tolerance, fps	0.121	0.183	0.189
Acceleration, Maximum, g's (max. thrust)	0.394	0.598	0.616
Acceleration, Minimum, g's (min. thrust)	0.0369	0.0399	0.0597

Propulsion Module Performance		
Maneuver	Propellant Settling & Thrust Vector Control	Midcourse, Orbit Insertion and Orbit Trim
Propellant	N_2H_4	N_2O_4 /Aerozine 50
Specific Impulse	(3)230 avg.	*
Total Impulse, lb-sec	38,200	*
Minimum Impulse Bit, lb-sec	2.0 $\pm$ .2	248 (est)
Start Impulse Tolerance, lb-sec	0.1	$\pm$100
Shutdown Impulse Tolerance, lb-sec	0.1	$\pm$100
Number of Engines	4	1
Thrust per Engine, lb	100	10,500 to 1050
Engine Operating Time, sec (each)	98	319 (max. thrust)
Thrust Vector Control	Pulse Mode	Pulse Monopropellant System
Subsystem Inert Weight, lb	243	3244
Propellant Weight, lb	166	11,347
Subsystem Weight, lb	409	14,591
Total Module Weight, lb	15,000	
Module Length, in.	150	

(1) With Minimum Thrust Main Engines and Propellant Settling Engines
(2) With Propellant settling Engines Only
(3) Corrected for Pulse Modes

*See D2-82709-10 Classified Supplement - Reference Page 15

Table 2.3-9: 1975 & 1977 LEM MODIFIED DESCENT PROPULSION MODULE,

SYSTEM PERFORMANCE (8,000-POUND CAPSULE)

System Performance	Midcourse	Orbit Insertion	Orbit Trim
Maximum Velocity, fps	656	4484	328
(1)Minimum Velocity, fps	3.22	Not Applicable	5.50
(2)Minimum Velocity, fps	.012	Not Applicable	.020
Maximum Velocity Tolerance, fps	.130	.208	.214
Acceleration, Maximum, g's (max. thrust)	.424	.676	.699
Acceleration, Minimum, g's (min. thrust)	.0396	.0430	.0676

Propulsion Module Performance		
Maneuver	Propellant Settling & Thrust Vector Control	Midcourse, Orbit Insertion & Orbit Trim
Propellant	N_2H_4	N_2O_4 /Aerozine 50
Specific Impulse	(3)230 avg.	*
Total Impulse, lb-sec	38,200	*
Minimum Impulse Bit, lb-sec	2.0 ± .2	248 (est)
Start Impulse Tolerance, lb-sec	.1	±100
Shutdown Impulse Tolerance, lb-sec	.1	±100
Number of Engines	4	1
Thrust per Engine, lb	100	10,500 - 1,050
Engine Operating Time, sec (each)	98	319 (at max thrust)
Thrust Vector Control	Pulse Mode	Pulse Monopropellant System
Subsystem Inert Weight, lb	243	3,244
Propellant Weight, lb	166	11,347
Subsystem Weight, lb	409	14,591
Total Module Weight, lb	15,000	
Module Length, in	150	

(1) With minimum thrust main engine and propellant settling engines
(2) With propellant settling engines only
(3) Corrected For Pulse Modes

*See D2-82709-10 Classified Supplement - Reference Page 16

Table 2.3-10: 1975 And 1977 LEM PROPULSION SYSTEM WEIGHT+ SUMMARY

Capsule	8000 and 10000
Allocated Bus and Science	3500
Engine System	405
*Propellant Supply System	510
*Pressurization System	530
Structure	1120
*Adapter Fittings	120
*Trapped Propellant	165
*Roll Control	31
*Additional Thermal Protection	105
*Instrumentation Sensors & Wiring	92
*Monopropellant	166
*Monopropellant System Inerts	243
Bipropellant Leakage	282
Usable Bipropellant	11,065
**Contingency	166
Planetary Vehicle Gross Weight	26,500 and 28,500 lb

* LEM Modifications

* * Contingency Includes a 3 percent Allowance for Weight Growth of Developed Hardware and a 10 percent Allowance for New Hardware.

\+ All weights expressed in pounds

2.4 OPTIMUM TRANSTAGE

The transtage consists of a propulsion module and a control module. For Voyager applications, the control module is removed, resulting in the configuration depicted in Figure 2.4-1. It is 120 inches in diameter and 167 inches in length. The forward end of the modified propulsion module (Station 167) is the Voyager payload interface. The modified transtage contains: (1) The main propulsion system for midcourse correction Mars orbit insertion, and Mars orbit trim; and (2) The secondary propulsion system for propellant settling.

A schematic of this propulsion system as modified for the Voyager mission is shown in Figure 2.4-2. The two unmodified pressure-fed fixed-thrust AJ10-138 rocket engines produce a combined vacuum thrust of 16,000 pounds. Both engine assemblies are mounted on a common frame. The component list is shown in Table 2.4-1. The existing propellant tanks are shortened by 20 inches and offloaded to meet the 15,000 pound gross weight allocation.

Both propellant tanks are equipped with a trap for multiple zero-gravity restarts and a screen to prevent bubbles from entering the outlets.

A feed line connects the tank outlet to the engine interface. Both tanks are pressurized by regulated helium gas stored in two high-pressure vessels.

The transtage multistart capability was originally limited to two restarts. However, in its R and D flight program, the transtage is scheduled to be started 10 times on one mission. It is therefore assumed that five restarts could be performed for the 1971 and 1973 Mars missions.

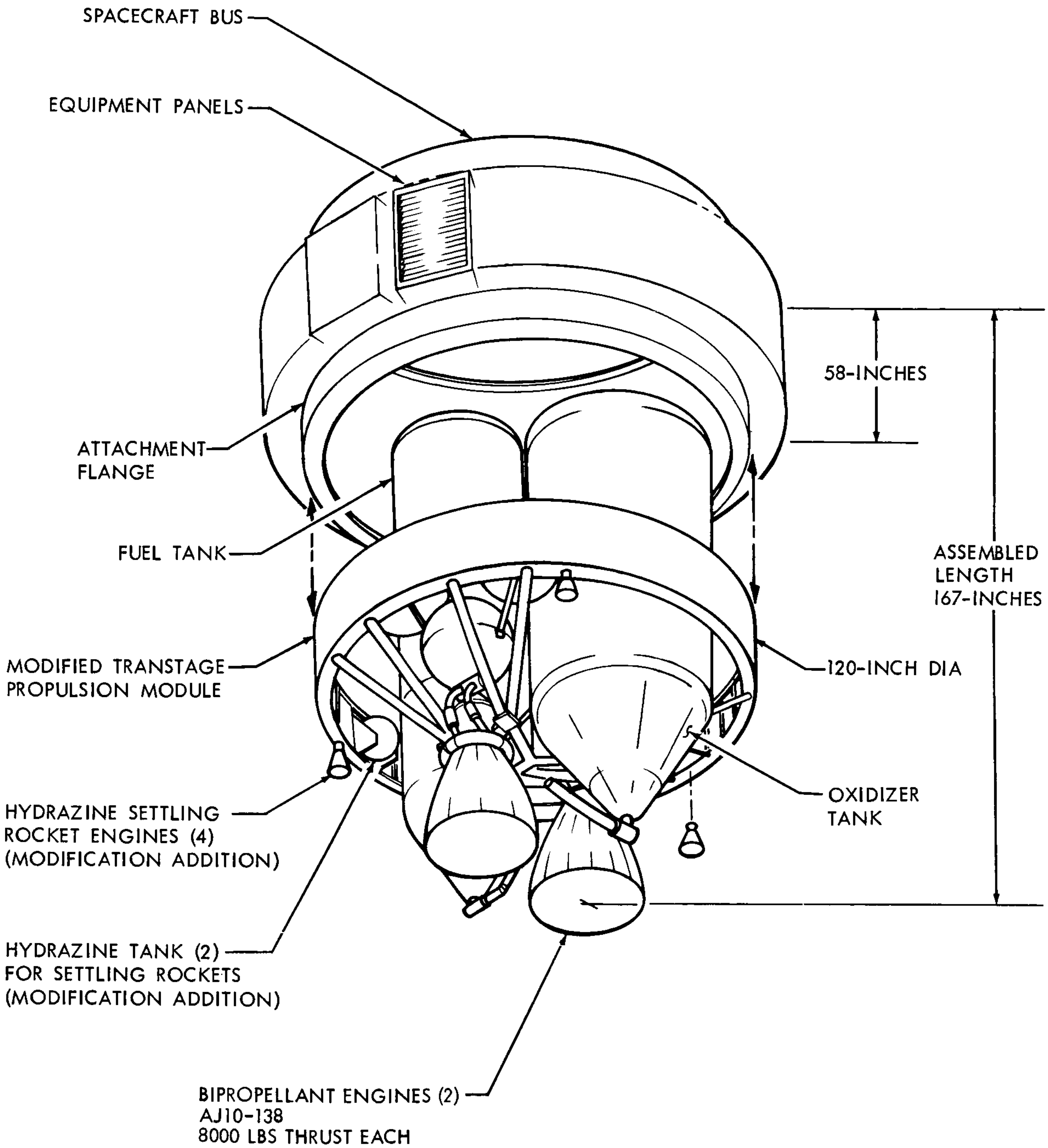
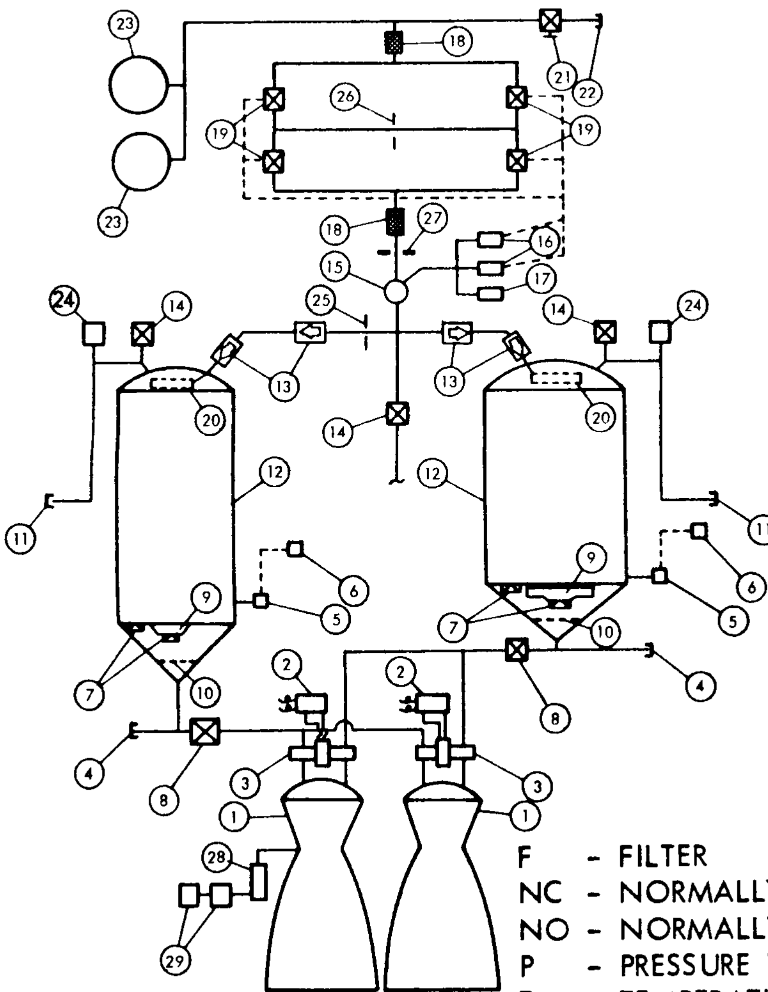
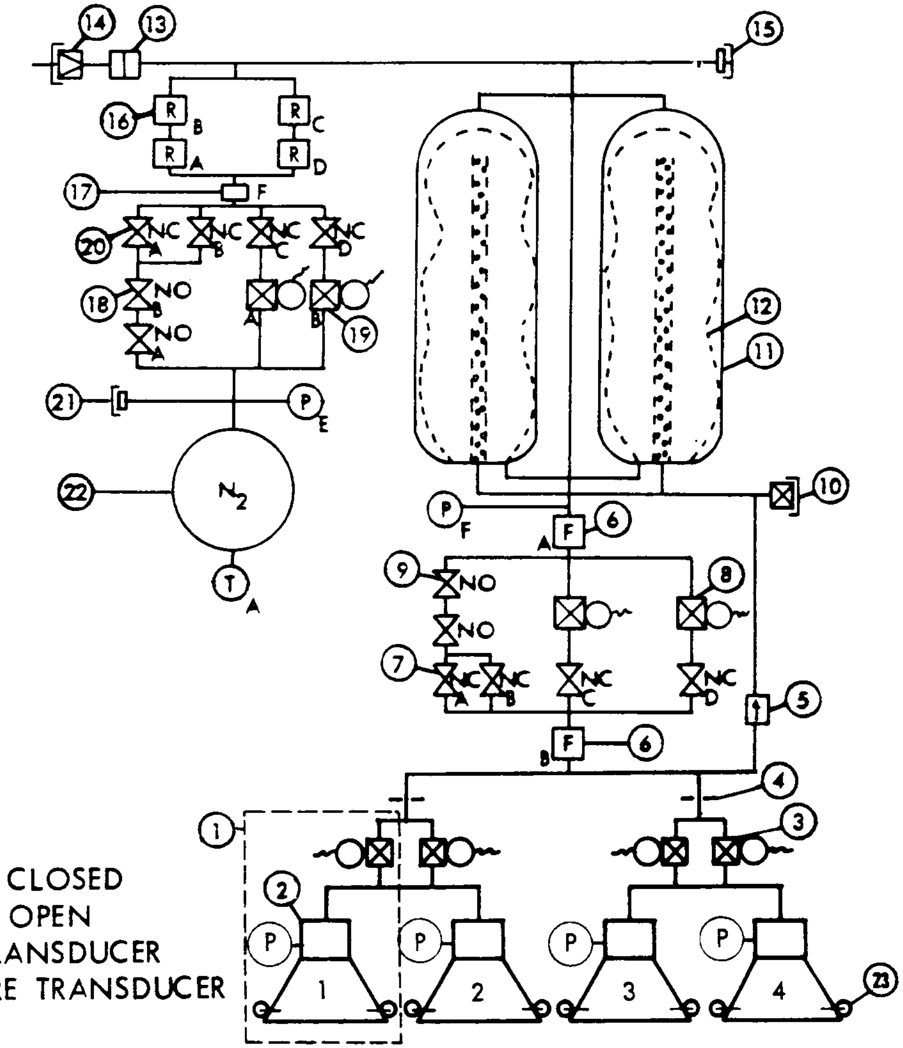


Figure 2.4-1: Modified Shortened Titan Transtage For Voyager Application

MODIFIED TRANSTAGE MAIN PROPULSION SYSTEM SCHEMATIC



MONOPROPELLANT SETTLING SYSTEM



F - FILTER
NC - NORMALLY CLOSED
NO - NORMALLY OPEN
P - PRESSURE TRANSDUCER
T - TEMPERATURE TRANSDUCER
R - REGULATOR

Figure 2.4-2: Shortened Transtage Propulsion System

Table 2.4-1: MODIFIED TRANSTAGE PROPULSION SYSTEM COMPONENT LIST

MAIN PROPULSION SYSTEM

ITEM	QTY.	NAME
1	2	Engine Nozzle & Thrust Chamber
2	2	Engine Pilot Valves & Solenoids
3	2	Bipropellant Valves
4	2	Propellant fill & Drain Connector & Cap
5	8	Outage Level Sensors
6	2	Controller
7	4	Check Valves
8	2	Prevalve
9	2	Baffle
10	2	Screen
11	2	Vent Coupling and Cap
12	2	Tank, Propellant
13	4	Check Valve
14	3	Relief Valve, Ordnance Operated
15	1	Accumulator
16	2	Pressure Switch, Solenoid Valve
17	1	Pressure Switch, Ground Check
18	2	Filter, Pressurizing Gas
19	4	Solenoid Valve
20	2	Diffuser, Pressurizing Gas
21	1	Manual Shut-Off Valve
22	1	Loading Connector and Cap
23	2	Tank, Pressurization Gas Storage
24	2	Pressure Switch, readiness monitor
25	1	Orifice, pressure balancing
26	1	Orifice, flow metering
27	1	Orifice, bleed
28	4	Actuators, Thrust Vector Control
29	1	Electric Motor and Hydraulic Pump
30		Tank Support Structure
31		Engine Support and Thrust Structure
32		Meteoroid Shield
33		Compartment and Component Heaters

Table 2.4-1 (Cont.)

MODIFIED TRANSTAGE PROPULSION SYSTEM COMPONENT LIST

MONOPROPELLANT SETTLING PROPULSION SYSTEM

ITEM	QTY.	NAME
1	4	Rocket Engine Assembly
2	4	Catalyst Bed
3	4	Valve, Latching Solenoid
4	2	Orifice
5	1	Thermal Relief
6	2	Filter, Propellant
7	4	Valve, Squib, N.C.
8	2	Valve, Latching Solenoid
9	2	Valve, Squib, N.O.
10	1	Valve and Cap, Fill and Drain
11	2	Tank, Propellant
12	2	Bladders, Positive Expulsion
13	1	Burst Disk
14	1	Valve, Relief
15	1	Valve and Cap, Vent and Press.
16	4	Regulator, Nitrogen
17	1	Filter, Nitrogen
18	2	Valve, Squib, N.O.
19	2	Valve, Latching Solenoid
20	4	Valve, Squib, N.C.
21	1	Valve and Cap, Press.
22	1	Tank, Nitrogen
23	16	Jet Vane and Actuator Assembly

Positive propellant positioning is accomplished by a separate monopropellant propulsion subsystem that consists of four 50-pound-thrust rocket engines with jet vane assemblies for thrust vector control.

Voyager spacecraft performance using the modified transtage propulsion module is summarized in Tables 2.4-2 for a 3000-pound capsule and 2.4-3 for a 2000-pound capsule.

The main propulsion engines and the settling engine performance characteristics are shown in Tables 2.4-4 and 2.4-5. The propulsion module weight statement is shown in Table 2.4-6. A summary of the transtage system reliability study is shown in Table 2.4-7.

For the 1975 and 1977 missions, hydrazine for settling the transtage propulsion system is increased from 105 to 122 pounds and bipropellant weight is decreased accordingly. System performance in 1975 and 1977 is shown in Tables 2.4-8 and 2.4-9 for 10,000- and 8,000- pound capsules,respectively

Detailed weights are given in Table 2.4-10.

The reliability summary presented for the 1971 and 1973 transtage system also applies for the 1975 and 1977 mission because the system components and operating conditions are similar. Transtage system reliability is 0.9907.

Table 2.4-2: 1971 & 1973 SHORTENED TRANSTAGE PROPULSION MODULE

SYSTEM PERFORMANCE (3000-POUND CAPSULE)

	SYSTEM PERFORMANCE	MIDCOURSE	ORBIT INSERTION	ORBIT TRIM
	Maximum Velocity, fps	656	7101	328
(1)	Minimum Velocity, fps	17.30 ± 2.98	Not Applicable	39.3 ± 6.54
(2)	Minimum Velocity, fps	.0031	Not Applicable	0.0072
	Maximum Velocity Tolerance, fps	1.85	3.93	4.06
	Acceleration, Maximum, g's	0.836	1.78	1.84
	Acceleration, Minimum, g's	0.782	0.849	1.78

PROPULSION MODULE PERFORMANCE

Maneuver	Prop. Settling and Attitude Control Assist	Midcourse, Orbit Insertion and Orbit Trim
Propellant	N_2H_4	N_2O_4/Aerozine 50
Specific Impulse, sec	235	*
Total Impulse, lb-sec	24,700	*
Minimum Impulse Bit, lb-sec (per Eng)	0.5	3,320 ± 885
Start Impulse Tolerance, lb-sec (per Eng)		± 334
Start Differential Impulse, lb-sec		300 to 90 percent thrust
Shutdown Impulse Tolerance, lb-sec (per Eng)		± 550
Shutdown Differential Impulse, lb-sec		212 to 5 percent thrust
Number of Engines	4	2
Thrust per Engine, lb	50	8000
Engine Operating Time, sec (each)	123	218
Thrust Vector Control	Jet Vanes	Gimbal
Subsystem Inert Weight, lb	220	2956
Propellant Weight, lb	105	11,719
Subsystem Weight, lb	325	14,675
Total Module Weight, lb	15,000	
Module Length, in.	167	

(1) With Main Engines and Propellant Settling Engines
(2) With Propellant Settling Engines Only

*See D2-82709-10 Classified Supplement - Reference Page 17

Table 2.4-3: 1975 & 1977 SHORTENED TRANSTAGE PROPULSION MODULE

SYSTEM PERFORMANCE (2000-POUND CAPSULE)

	SYSTEM PERFORMANCE	MIDCOURSE	ORBIT INSERTION	ORBIT TRIM
	Maximum Velocity, fps	656	7783	328
(1)	Minimum Velocity, fps	18.2 ± 3.13	Not Applicable	44.4 ± 7.39
(2)	Minimum Velocity, fps	0.0033	Not Applicable	.0081
	Maximum Velocity Tolerance, fps	1.95	4.44	4.59
	Acceleration, Maximum, g's	0.879	2.01	2.07
	Acceleration, Minimum, g's	0.822	.893	2.01

PROPULSION MODULE PERFORMANCE

Maneuver	Propellant Settling and Attitude Control Assist	Midcourse, Orbit Insertion and Orbit Trim
Propellant	N_2H_4	N_2O_4/Aerozine 50
Specific Impulse, sec	235	*
Total Impulse, lb-sec	24,700	*
Minimum Impulse bit, (per Eng) lb-sec	0.5	3320 + 885
Start Impulse Tolerance, (per Eng) lb-sec		± 334
Start Differential Impulse, lb-sec		300 to 90 percent thrust
Shutdown Impulse Tolerance, lb-sec		± 550
Shutdown Differential Impulse, lb-sec		212 to 5 percent thrust
Number of Engines	4	2
Thrust per Engine, lb	50	8000
Engine Operating Time, sec (each)	123	218
Thrust Vector Control	Jet Vanes	Gimbal
Subsystem Inert Weight, lb	220	2956
Propellant Weight, lb	105	11,719
Subsystem Weight, lb	325	14,675
Total Module Weight. lb	15,000	
Module Length, in.	167	

(1) With Main Engines and Propellant Settling Engines
(2) With Propellant Settling Engines Only

*See D2-82709-10 Classified Supplement - Reference Page 18

Table 2.4-4: SHORTENED TRANSTAGE PROPULSION MODULE--MAIN ENGINE DATA SHEET

Designation	AJ10-138
Manufacturer	Aerojet-General
Number of Engines	2
Status	Flight Test
Propellants: Fuel	Aerozine-50
Oxidizer	N_2O_4
Vacuum Thrust per Engine	8000 lb
Engine Specific Impulse	* sec (Spec.Min)
Mixture Ratio O/F	2.0 $\pm$ 0.04
Expansion Area Ratio	40
Chamber Pressure	105 $\pm$ 5 psia
Engine Start and Shutdown Transients:	
T/C - Valve Opening Time	*
T/C - Valve Closing Time	*
Start Transient Impulse	*
Shutdown Transient Impulse	*
Start Differential Impulse	*
Shutdown Differential Impulse	*
Total Impulse for Minimum Pulse Width	*
Throttle Ratio	None
Restart Capability	Multi
Total Mission Burn Time	218 sec
Ignition	Hypergolic
Cooling	Ablative with radiation skirt
Weight, Dry	175 lb
Size: Length	81 in
Diameter (maximum)	47.5 in
Thrust Vector: Type	Gimbal
Angle	$\pm$ 9 degrees (combined pitch and yaw)
Rate	50 degrees/sec
Acceleration	403 degrees/sec^2
Fuel Inlet Pressure (Average)	161 psia
Oxidizer Inlet Pressure (Average)	163 psia

*See D2-82709-10 Classified Supplement - Reference 19

Table 2.4-5: SHORTENED TRANSTAGE PROPULSION MODULE, PROPELLANT SETTLING--
ENGINE DATA SHEET

Designation	
Manufacturer	
Status	New Development Required
Propellants	Hydrazine Monopropellant
Engine Thrust, Vacuum	50 LBF
Engine Specific Impulse, Vac.	235 sec
Expansion Ratio A_e/A_t	50
Exit Area	9.40 sq. in.
Chamber Pressure	150 psia
Start Time Impulse	20 ms ON to first rise cold 50 ms first rise to 90 percent cold
Shutdown Time and Impulse	10 ms OFF to first drop 100 ms first drop to 10 percent
Minimum Total Impulse Bit	12.5 lb-sec
Throttle Ratio	None
Restart Capability	Multiple
Burn Time or Service Life	1000 sec
Ignition	Spontaneous catalyst
Cooling	Radiation
Weight, Dry	2.5 lb 2.6 lb for jet vanes
Size: Length	5.7 in
Diameter	3.5 in
Thrust Vector: Type	Jet vanes
Angle	5 degrees effective
Fuel Input Pressure	260 psi nominal

Table 2.4-6: TRANSTAGE PROPULSION PLANETARY VEHICLE WEIGHT SUMMARY

	Shortened
Capsule	2000 and 3000
Allocated Bus and Science	2500
Propulsion Installation	(15,000)
Body Group (Structure)	169
Separation and Destruct	88
Propulsion (Engines)	410
*Propulsion (Tankage and Feed System)	1390
Orientation Control (Tank Baffles and Screens)	78
Pressurant (Helium)	45
Trapped Propellant	100
*Propellant Tank Factor of Safety Increase	220
*Instrumentation, Sensors and Wiring	92
*Meteoroid Shielding	84
*Thermal Protection	198
*Meteoroid/Thermal Support Structure	92
*Propellant Leakage	282
*Pressurization Tank Gage Decrease	-150
*Mono-Propellant System Inerts	220
*Usable Monopropellant	105
Usable Bipropellant	11,437
**Contingency	140
Planetary Vehicle Gross Wt.	19,500 and 20,500 lb

NOTES: *Transtage Modifications
**Contingency includes a 3 percent allowance for weight growth of developed hardware and a 10 percent allowance for new hardware.

Table 2.4-7: 1971 and 1973 CONFIGURATION TITAN III TRANSTAGE PROPULSION SYSTEM RELIABILITY SUMMARY

Item	$\lambda \times 10^6$	N	t	Failures/10^6 $\lambda Nt \times 10^6$
TRANSTAGE ENGINE				
Piping, Tanks and Connections	0.167/hr	1	5112 hrs	855
Valve, Solenoid	2/cy.	Redun.	1 cy.	12
Filter	3.3/hr	2	0.3 hr	2
Pressure Switch	36/hr	Redun.	0.3 hr	Negl.
Accumulator	0.08/hr	1	0.3 hr	Negl.
Check Valves	1/hr	Redun.	0.3 hr	Negl.
Relief Valves - Closed	1/cy.	3 Redun.	5 cy.	Negl.
- Open	1/cy.	3	5 cy.	15
Tank, Fuel and Oxidizer	0.0803/hr	2	5112 hrs	819
Pilot Valve, Solenoid-Closed	1/cy.	2	5 cy.	10
-Open		Redun.	5 cy.	Negl.
Valve, Bipropellant-Closed	25/cy.	2	5 cy.	250
Pressure Operated-Open	25/cy.	Redun.	5 cy.	Negl.
Engine, Bipropellant	200/cy.	2	5 cy.	2000
Gimbal	2.5/hr	2	0.1 hr	Negl.
Actuator, Gimbal	3/hr	4	0.1 hr	2
		Total		3965
		Re = 0.9960		
SETTLING SYSTEM				
Monopropellant System	0.268/hr	1 syst.	5112 Hrs	1370
4 Engines (no engine out)	100/cy.	4	10 cy.	4000
		Total		5370
		Re = 0.9946		
Transtage plus Monopropellant System		Re = 0.9907		

λ = Failures/Hr or Cycle
N = Number of Components
t = Hours or Cycles

See note at conclusion of Paragraph 7.4.

Table 2.4-8: 1975 & 1977 SHORTENED TRANSTAGE PROPULSION MODULE SYSTEM PERFORMANCE

(10,000-POUND CAPSULE)

SYSTEM PERFORMANCE	MIDCOURSE	ORBIT INSERTION	ORBIT TRIM
Maximum Velocity, fps	656	4027	328
(1) Minimum Velocity, fps	13.3 ± 2.01	Not Applicable	21.9 ± 3.41
(2) Minimum Velocity, fps	0.0023	Not Applicable	0.0037
Maximum Velocity Tolerance, fps	1.33	2.05	2.12
Acceleration, Maximum, g's	0.601	0.925	0.955
Acceleration, Minimum, g's	0.562	0.608	0.925

PROPULSION MODULE PERFORMANCE		
Maneuver	Propellant Settling and Attitude Control Assist	Midcourse Orbit Insertion and Orbit Trim
Propellant	N_2H_4	N_2O_4/Aerozine 50
Specific Impulse, sec	235	*
Total Impulse, lb-sec	28,700	*
Minimum Impulse Bit, (per Eng) lb-sec	0.5	3,320 ± 885
Start Impulse Tolerance, (per Eng) lb-sec		± 334
Start Differential Impulse, lb-sec		300 to 90 percent thrust
Shutdown Impulse Tolerance,(per Eng) lb-sec		± 550
Shutdown Differential Impulse		212 to 5 percent thrust
Number of Engines	4	2
Thrust per Engine, lb	50	8000
Engine Operating Time, sec (each)	143	217
Thrust Vector Control	Jet Vanes	Gimbal
Subsystem Inert Weight, lb	235	2956
Propellant Weight, lb	122	11,687
Subsystem Weight, lb	357	14,643
Total Module Weight, lb	15,000	
Module Length, in.	167	

(1) With Main Engines and Propellant Settling Engines
(2) With Propellant Settling Engines Only

*See D2-82709-10 Classified Supplement - Reference Page 20

Table 2.4- 9: 1975 & 1977 SHORTENED TRANSTAGE PROPULSION MODULE SYSTEM PERFORMANCE

(8000-POUND CAPSULE)

SYSTEM PERFORMANCE	MIDCOURSE	ORBIT INSERTION	ORIBT TRIM
Maximum Velocity, fps	656	4559	328
(1) Minimum Velocity, fps	14.3 ± 2.16	Not Applicable	24.8 ± 3.86
(2) Minimum Velocity, fps	0.0024	Not Applicable	0.0042
Maximum Velocity Tolerance, fps	1.43	2.32	2.40
Acceleration, Maximum, g's	0.646	1.05	1.08
Acceleration, Minimum, g's	0.605	0.655	1.05

PROPULSION MODULE PERFORMANCE		
Maneuver	Propellant Settling and Attitude Control Assist	Midcourse Orbit Insertion and Orbit Trim
Propellant	N_2H_4	N_2O_4/Aerozine 50
Specific Impulse, sec	235	*
Total Impulse, lb-sec	28,700	*
Minimum Impulse Bit, (per Eng) lb-sec	0.5	3,320 ± 885
Start Impulse Tolerance, (per Eng) lb-sec		± 334
Start Differential Impulse, lb-sec		300 to 90 percent thrust
Shutdown Impulse Tolerance, (per Eng) lb-sec		± 550
Shutdown Differential Impulse, lb-sec		212 to 5 percent thrust
Number of Engines	4	2
Thrust per Engine, lb	50	8000
Engine Operating Time, sec (each)	143	217
Thrust Vector Control	Jet Vanes	Gimbal
Subsystem Inert Weight, lb	235	2956
Propellant Weight, lb	122	11,687
Subsystem Weight, lb	357	14,643
Total Module Weight, lb	15,000	
Module Length, in.	167	

(1) With Main Engines and Propellant Settling Engines
(2) With Propellant Settling Engines Only

*See D2-82709-10 Classified Supplement - Reference Page 21

Table 2.4-10: TRANSTAGE PROPULSION PLANETARY VEHICLE WEIGHT SUMMARY 1975 and 1977 MISSIONS

	Shortened
Capsule	8000 and 10,000 lb
Allocated Bus and Science	3500
Propulsion Installation	(15,000)
Body Group (Structure)	169
Separation and Destruct	88
Propulsion (Engines)	410
* Propulsion (Tankage and Feed System)	1390
Orientation Control (Tank Baffles and Screens)	78
Pressurant (Helium)	45
Trapped Propellant	100
*Propellant Tank Factor of Safety Increase	220
*Instrumentation, Sensors and Wiring	92
*Meteoroid Shielding	84
*Thermal Protection	198
*Meteoroid/Thermal Support Structure	92
*Propellant Leakage	282
*Pressurization Tank Gage Decrease	-150
*Mono-Propellant System Inerts	235
*Usable Monopropellant	122
Usable Bipropellant	11,405
**Contingency	140
PLANETARY VEHICLE GROSS WEIGHT	26,500 and 28,500 lb

NOTES: *Transtage Modifications
**Contingency includes a 3 percent allowance for weight growth of developed hardware and a 10 percent allowance for new hardware.

3.0 PREFERRED DESIGN SELECTION

3.1 COMPETING CHARACTERISTICS

Preferred-design selection is based on the following competing characteristics, (as given by JPL) in order of decreasing priority:

1) Probability of mission success;
2) Performance of mission objectives;
3) Cost savings;
4) Contributions to subsequent missions;
5) Additional 1971 capability.

These competing characteristics are discussed below for candidate propulsion systems. A preferred design selection follows in Section 3.2.

3.1.1 Probability of Mission Success

Mission success probability is influenced by the following key factors:

1) Extent of modifications required to adapt existing propulsion system hardware and technology to Voyager requirements;
2) Predicted reliability of propulsion system;
3) Compatibility of propulsion system hardware with Voyager mission environment;
4) Compatibility of propulsion system with planetary and space vehicles;
5) Compatibility of propulsion system with planetary quarantine requirements.

3.1.1.1 Extent of Modifications

The modifications required to adapt the Minuteman motor and Mariner IV hydrazine subsystem into an optimum solid/monopropellant Voyager propulsion concept are summarized in Table 3.1-1.

TABLE 3.1-1: SOLID/MONO CONCEPT MODIFICATIONS

MONOPROPELLANT SYSTEM			SOLID PROPELLANT SYSTEM		
VOYAGER	MARINER IV	COMMENT	VOYAGER	MINUTEMAN	COMMENT
Multiple Engines (N_2H_4)	Single Engine (N_2H_4)	Variable CG, Engine-Out	One Shortened Motor	One Motor	Propellant Loading Requirement
Spontaneous Catalyst	Hypergolic Start Slugs	Multiple Re-start; Reliability	N_2 for Pressurant and Roll Control	Hot Gas Gen.	Vehicle Compatibility
Solenoid Engine Valves, Latching	Remote Squib Valves	Multiple Restart Consistent start and cutoff Transients	Regulator (4)	Pres. Relief Valve	System Weight
Cylindrical N_2H_4 Tanks, Butyl Bladder	Spherical N_2H_4 Tanks, Butyl Bladder	Packaging	Quad Sol. Valve (2)	Equal. Valve	Sys. Comp.
Multiple N_2 Tanks	One N_2 Tank	C.G. Control	Lengthen Nozzle 15"	Existing Nozzle	Panel Heating
N_2 Solenoid & Squib Valves	Squib Valves	Multiple Positive Isolation	Add Nozzle Seal	None	Space Storage
N_2H_4 Sol & Squib Valves	Squib Valves	Multiple Positive Isolation	Titanium Freon Tank	Stainless Steel Freon Tank	Magnetics
Four N_2 Regulators	Single Regulator	Regulator-Out Capability	Freon Operated Hydraulic Valves	Hydraulic Oil and Pumps	Power
Burst Disk, N_2	None	Over-Pressure Protection			
Relief Valve, N_2	**None**	**Over-Pressure Protection**			
Thermal Relief Orifice, N_2H_4	None	Over-Pressure Protection			

Significant solid/monopropellant system modifications to Minuteman and Mariner propulsion are in the areas of solid-motor thrust vector control assembly, motor and nozzle lengths, and hydrazine engine catalyst. LEM descent propulsion modifications are listed in Table 3.1-2. Key modifications to the LEM descent propulsion system are in the areas of thrust vector control, propellant settling, helium storage and thermal control.

Titan III C transtage modifications are listed in Table 3.1-3. Key transtage modifications are in the areas of propellant settling, shortened tankage, leakage minimization, micrometeoroid shielding, and thermal control.

It is concluded that the required modifications to both Minuteman solid/hydrazine subsystem concepts and the LEM descent propulsion system are similar in scope. Transtage modifications are extensive by comparison.

3.1.1.2 Reliability

Predicted reliability numbers for the three competing concepts as a function of mission time are shown in Figure 3.1-1. It is assumed that all three concepts will have achieved their mature reliability values by 1971.

It is concluded that the solid/monopropellant system, with a reliability of 0.996, is slightly more reliable than the bipropellant stages. This is because the highly reliable solid motor subsystem of this concept is used only once (for orbit insertion). By contrast, the two bipropellant stages must perform orbit trim maneuvers after orbit insertion. The LEM descent propulsion system, with a reliability of 0.990, is less reliable

TABLE 3.1-2: LEM PROPULSION SYSTEM MODIFICATIONS

MONOPROPELLANT SETTLING SYSTEM				MAIN PROPULSION SYSTEM			
ITEM	VOYAGER	MARINER	COMMENT	ITEM	VOYAGER	LEM	COMMENT
Propellant Expulsion Device	Butyl Bladder	Butyl Bladder	Mariner Experience	Pressurization Gas Storage	Ambient Temperature Helium	Cryogenic Helium	Longer Mission Time
Propellant Tank	Cylindrical	Spherical	Packaging	Landing Gear	None	Outrigger Legs	No Requirement
Catalyst	Shell 405 Spontaneous	Hypergolic Slug + Non-spontaneous Catalyst	Multiple Restart	Thermal Control	Semipassive	Passive	Longer Mission Time & Deeper-Space Application
Pressure Regulator	Quad	Single	Redundant Reliability				
Solenoid Latching	12 Valves	None	Redundant Reliability	Thrust Vector Control	Pulse Mode Operation of Mono-Propellant Settling Rocket Engines (Main Engine Locked	Electro-Mechanical Actuators	Vehicle C.G. Excursions
Propellant Filter	Two	None	Reliability				
Burst Disk	One	None	Leakage				
Thermal Relief Valve	One	None	Relieve Isolated Propellant				

TABLE 3.1-3: TRANSTAGE PROPULSION SYSTEM MODIFICATIONS

MONOPROPELLANT SETTLING SYSTEM			
ITEM	VOYAGER	MARINER	COMMENT
Propellant Expulsion Device	Bladder	Bladder	No Change
Propellant Tank	Cylindrical	Spherical	Packaging
Catalyst	Shell 405 Spontaneous	Hypergolic Slug + Non-spontaneous Catalyst	Multiple Restart
Pressure Regulator	Quad	Single	Redundant Reliability
Solenoid Latching Valves	12 Valves	None	Redundant Reliability
Propellant Filter	Two	None	Reliability
Burst Disk	One	None	Relief Valve Leakage
Thermal **Relief** Valve	One	None	Relieve Isolated Propellant

MAIN PROPULSION SYSTEM			
ITEM	VOYAGER	TRANSTAGE	COMMENT
Engine Propellant Valves	Redundant Squib Operated	Solenoid Pilot,Fuel Operated	Leakage Reliability
Meteoroid Shield	Aft Meteoroid Shield	None	Mars Mission Applications
Prevalves	Fuel & Oxidizer	None	Shutdown Redundancy
Relief Valves	Pressure System Only	None	Longer Mission
Thermal Control	Semipassive	Passive	Longer Mission Time & Space Applications
Pressure Vessels	Increase Gage		Increase Safety Factor
Joints	Change		Reduce Leakage

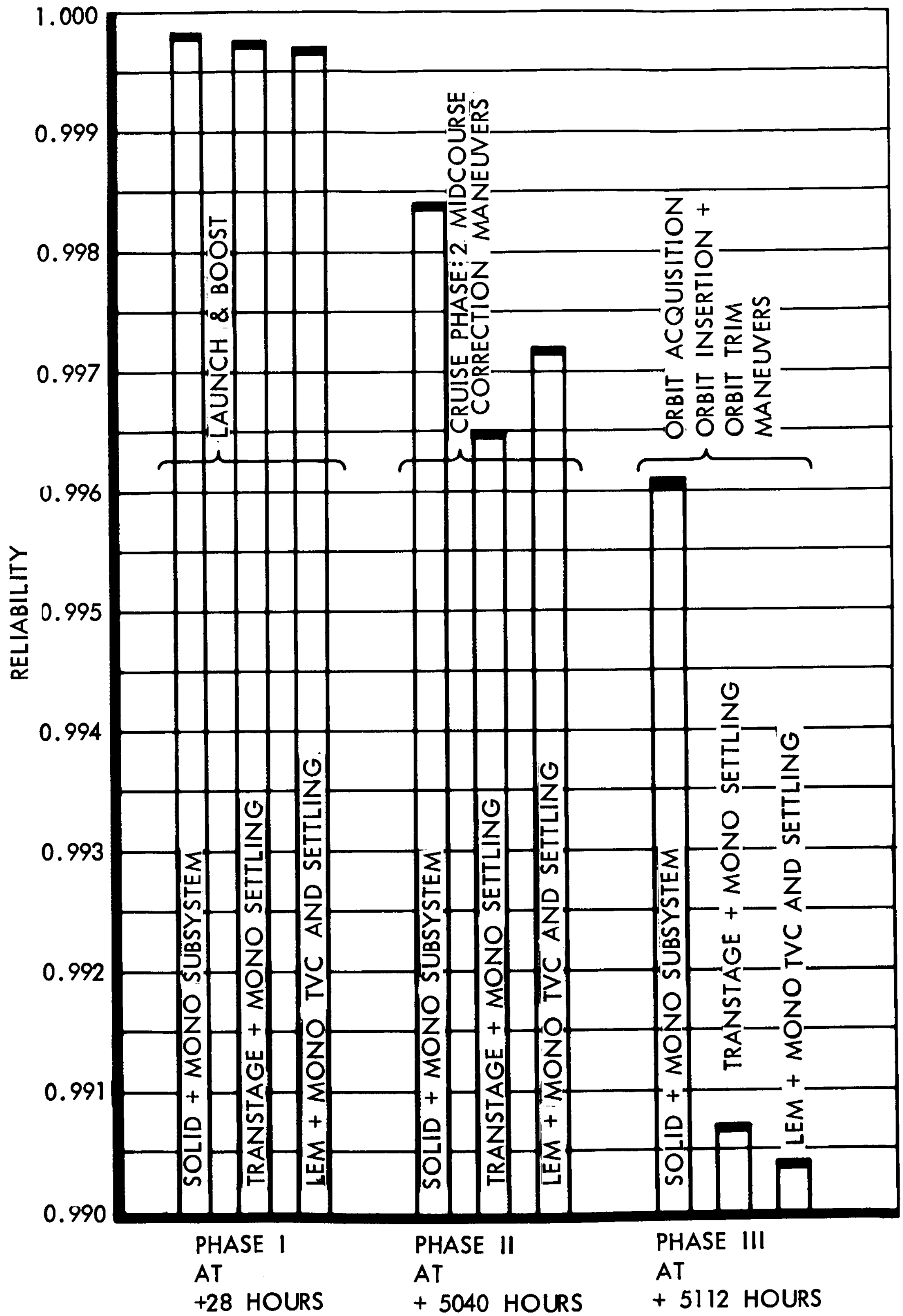


Figure 3.1-1: Propulsion System Trade Study — Comparative Reliabilities

than transtage (0.991) because of the complex thrust-vector control system required to adapt the LEM propulsion stage to the Voyager application.

3.1.1.3 Compatibility with Mission Environment

Natural Environment--Prolonged exposure to hard vacuum is the critical environmental consideration for Voyager propulsion. The current status of testing of candidate propulsion systems is summarized in Table 3.1-4.

It is concluded, on the basis of available test data and the Mariner IV experience, that the modified Minuteman motor and attendant Mariner-type hydrazine subsystem are compatible with the Voyager mission environment. In addition, the Minuteman motor has been fired successfully after prolonged ambient storage (in excess of 20 months); this indicates that propellant hardening due to aging is not detrimental to motor performance.

It is assumed that the two bipropellant stages can be made compatible with prolonged exposure to deep space. However, no substantiating data are currently available.

Induced Environment--The environment induced by the Space Vehicle during boost, and the environment induced by the propulsion system during propulsion maneuvers, must be considered as to their effects on the propulsion system.

1) Boost Environment--The Minuteman motor and transtage are designed to withstand the boost environments of the Minuteman and Titan III-C missiles, respectively. The LEM descent propulsion system is currently designed for launch by the Saturn V booster. It is therefore compatible with the boost environment.

TABLE 3.1-4: FLIGHT AND TEST DATA IN SPACE ENVIRONMENT

SYSTEM	SPACE EXPOSURE
SOLID/MONO	
MODIFIED MINUTEMAN PROPELLANT	60-DAY TEST IN VACUUM CHAMBER
MONOPROPELLANT HYDRAZINE	MARINER MARS MISSION (250 DAYS)
SHELL CATALYST	30-DAY TEST IN VACUUM CHAMBER
LEM	8-DAY LUNAR MISSION*
TRANSTAGE	6.5 - HOUR EARTH ORBITAL MISSION
* BY 1970	

It appears that the four competing concepts are compatible with the boost environment. The modified Minuteman motor, which is used in the two solid/liquid concepts, has an advantage because the boost environment of the silo-launched Minuteman missile is particularly severe.

2) Self-Induced Environment--The environment in which the modified Minuteman motor performs the Voyager mission is less severe than in the original Minuteman application. The shortened motor case results in lower chamber pressures and temperatures. This lowers both structural and thermal loads. Despite longer burn times, total heat loads to the motor case liner and nozzle are lower than those for the unmodified motor. The LEM descent propulsion system results in higher soak-back temperatures than the other three systems considered. In its original application in the lunar mission, the role of the LEM descent engine is completed after the main retro maneuver. Therefore, the effects of high (approximately 1000°F) soak-back temperatures on the LEM engine system do not affect the success of the Apollo mission. For the Voyager mission, however, the LEM descent engine must perform two orbit trim maneuvers after the main retro maneuver. The capability of the LEM descent system to perform after the main retro maneuver is not known at this time.

It is concluded that the two solid/liquid designs are more compatible with the mission environment than the two bipropellant stages.

3.1.1.4 Compatibility with Planetary and Space Vehicles

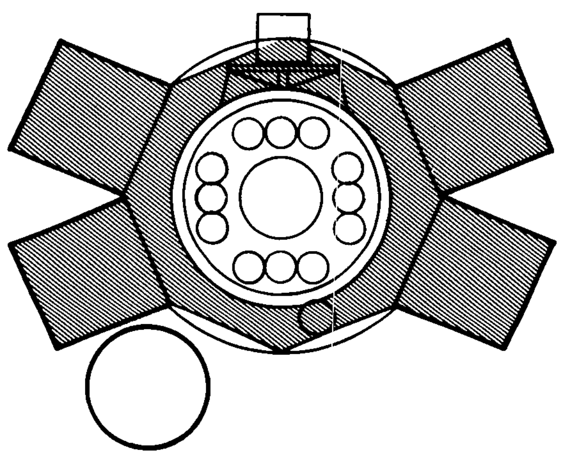
The two solid/hydrazine propulsion systems cause the highest inertial and solar-panel heating loads during orbit insertion. They are made compatible with the spacecraft by moderate structural strengthening of appendages and by a motor nozzle extension. The resulting slight weight increment is reflected in a reduction of orbit insertion velocity performance that could otherwise be obtained.

The diameter of the Spacecraft Bus associated with the solid/hydrazine system is 120 inches. The boost shroud inside diameter is 240 inches, allowing for mounting over 50 percent of the solar cells on fixed panels, as shown in Figure 3.1-2. This arrangement is advantageous compared to those involving totally deployed panels. Solar-panel area loss due to a single failure in hinge-deployment is 11.5 percent.

The LEM descent propulsion results in a Spacecraft Bus that is 180 inches wide. The wider propulsion module necessitates a larger percentage of deployable solar panels than the solid/liquid systems because of solar cell temperature considerations. A representative LEM/Spacecraft configuration is shown in Figure 3.1-2. Solar power loss due to a single failure in hinge-deployment can be as high as 33 percent for this configuration.

Consideration of thrust vector control requirements reveals a serious incompatibility of the unmodified LEM descent propulsion system with the Planetary Vehicle. In its original application, the distance between LEM vehicle center of gravity and descent engine trunnion point is in excess of 30 inches. The $\pm 6^{\circ}$ gimbaled LEM descent engine provides adequate thrust vector control in that condition. In the Voyager application, however, vehicle center of gravity is as close as 3 inches to the trunnion point as shown in Figure 3.1-3, and the center of gravity may shift, with capsule off, from one side of the trunnion point to the other during the mission. In addition, the LEM TVC actuators operate at the relatively low rate of 0.4°/sec.

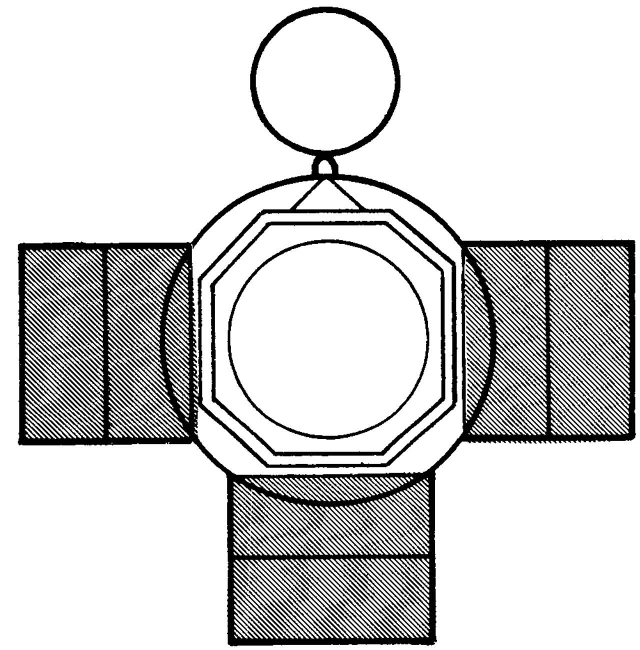
SOLID/LIQUID UNIT



SOLAR PANEL AREA
FIXED 178 SQ FT
DEPLOYABLE 152 SQ FT

11.5% LOSS FOR SINGLE DEPLOYMENT FAILURE

LEM



SOLAR PANEL AREA
FIXED 0 SQ FT
DEPLOYABLE 330 SQ FT

33% LOSS FOR SINGLE DEPLOYMENT FAILURE

Figure 3.1-2 Effect of Propulsion System on Solar Panel Design

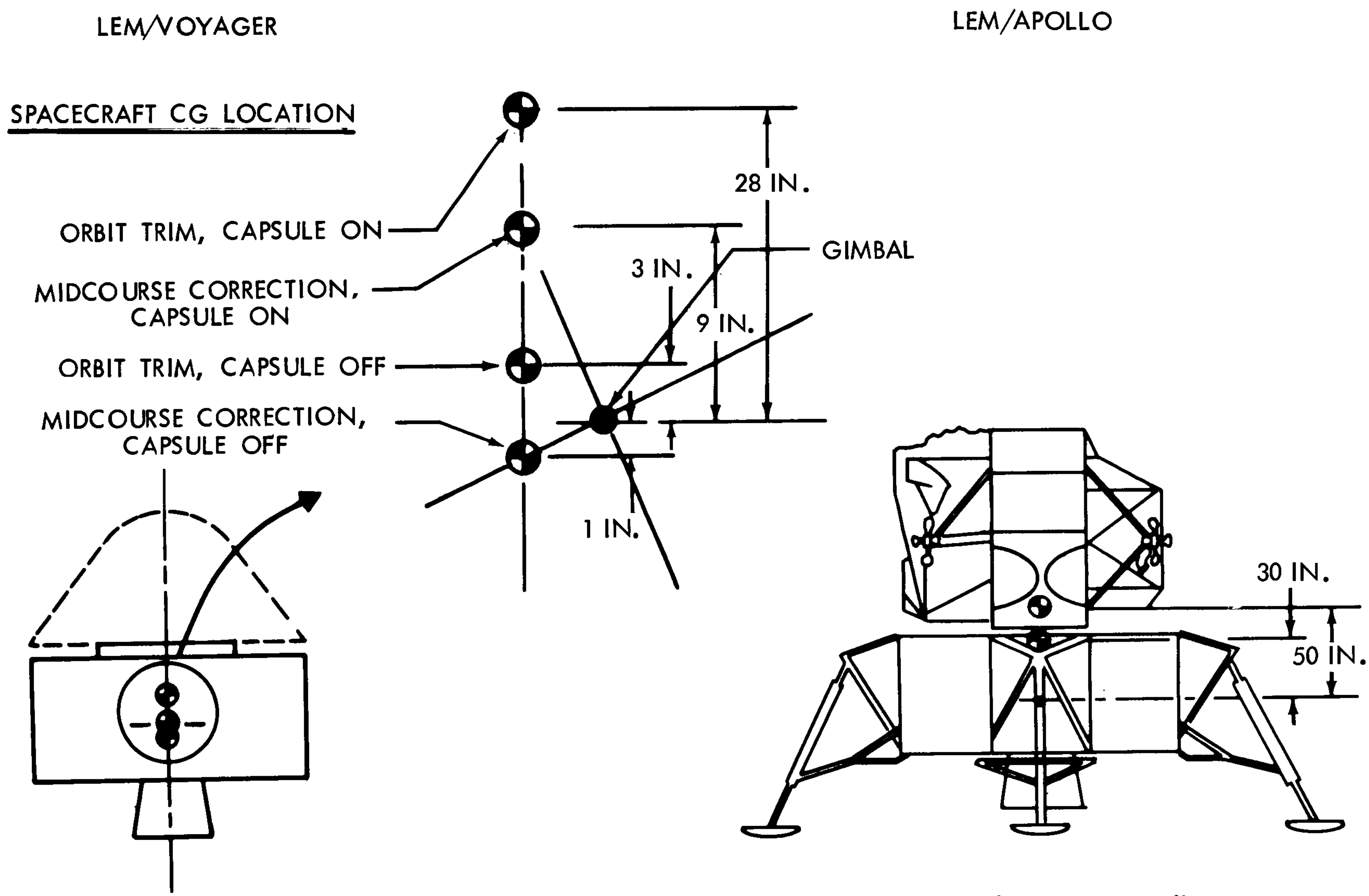


Figure 3.1-3: LEM Descent Propulsion System Compatability With Spacecraft — Thrust Vector Control

As a consequence, the gimbaled LEM engine cannot cope with Voyager thrust vector control requirements. To adapt this propulsion system to Voyager, the LEM descent engine is locked in position and a four-engine hydrazine system operating in a pulsed mode is added, at reliability and weight penalties, to provide the required thrust vector control.

The solid/hydrazine and LEM propulsion systems considered resulted in comparable Flight Spacecraft lengths under the shroud. The solid/hydrazine propulsion systems result in a 158-inch shroud length, compared with 150 inches for the LEM descent system. The shortened transtage resulted in a shroud length of 167 inches, considerably longer than the other three concepts.

It is concluded that the four propulsion systems have nearly the same impact on the planetary and space vehicle design. The solid/hydrazine concept has an advantage in that it allows for more than half the solar panel area to be fixed. The LEM concept has the advantage of resulting in the shortest spacecraft. The shortened transtage is least compatible with the space vehicle because its greater length results in higher booster aerodynamic loads than for the other concepts.

3.1.1.5 Compatibility with Planetary Quarantine

Planetary quarantine requirements result in an allocated probability of less than 4×10^{-6} that unsterile propulsion system ejecta will contaminate Mars. Components of the hydrazine subsystem of the solid/hydrazine concepts which are in contact with hydrazine are self-sterilizable. Available data indicate, with a reasonable probability, that a solid motor is self-sterilizing during web burn. The solid/liquid system is made compatible with planetary quarantine requirements by decontaminating the liquid subsystem pressurization system and TVC assembly surfaces exposed to nitrogen and freon with ethylene oxide, and loading it aseptically with filtered freon and nitrogen. In the bipropellant stages, the fuel is self-sterilizing and can be loaded directly. The self-sterility of the oxidizer is questionable on the basis of available data. The surfaces of the oxidizer and pressurization subsystem components that come in contact with N_2O_4 and nitrogen will be decontaminated with ethylene oxide. Filtered oxidizer and pressurant are then loaded aseptically.

It is concluded that the four candidate propulsion systems are equally compatible with planetary quarantine requirements.

3.1.2 Performance of Mission Objectives

In performing its mission objectives, the propulsion system must accomplish the following:

1) Delivery of design goal velocity increments for all propulsive maneuvers;
2) Delivery of velocity increments to high accuracy to meet trajectory and orbit accuracy requirements;
3) Insertion of the planetary vehicle into all orbits whose total impulse requirements are less than or equal to the design goal total impulse;
4) Delivery of minimum velocity increments for midcourse corrections and orbit trim.

The capability of the candidate propulsion systems to perform the above mission objectives is discussed below.

3.1.2.1 Velocity Performance

All propulsion concepts are required to provide 200 m/sec for trajectory correction and Planetary Vehicle separation, and 100 m/sec for orbit trim.

Their 1971 velocity performance for inserting the Planetary Vehicle (with a 2000-pound capsule) into a Mars orbit is summarized in Figure 3.1-4. The four propulsion systems meet the design goal of 2.2 Km/sec (7216 fps) for the 15,000-pound weight allocation. The modified Minuteman solid/hydrazine subsystem unit, sized for the 1971 and 1973 missions, provides the highest orbit insertion velocity increment of 2.47 Km/sec (8101 fps). The LEM descent propulsion system provides the lowest orbit insertion velocity increment of 2.31 Km/sec (7597 fps).

LEM system performance is based on a 2500-pound Flight Spacecraft. Because of its configuration and structural arrangement, the LEM descent propulsion system results in lower weights for the Spacecraft Bus, Planetary Vehicle Adapter, and boost shroud, as indicated in Table 3.1-5. If the LEM system were credited with additional propellant allowed by the above weight savings its orbit-insertion velocity performance would be as shown by point A on Figure 3.1-4. Unlike the LEM system, the transtage results in increased Spacecraft Bus and shroud weights. If penalized by a propellant weight required by the increased bus and shroud weights, its performance would be as shown by point B in Figure 3.1-4.

The 1971 orbit insertion capability of the four propulsion concepts, with a 3000-pound capsule, is shown in Figure 3.1-5. Only the modified Minuteman/hydrazine unit, sized for the 1971 and 1973 missions, exceeds the design goal of 2.2 Km/sec (7216 fps). The other three propulsion systems, however, exceed the required minimum velocity increment of 2.0 Km/sec (6560 fps). The modified Minuteman/hydrazine unit, sized for the 1975 and 1977 missions, provides the second highest orbit insertion velocity increment capability. It is within 24 m/sec (79 fps) of meeting the 2.2 Km/sec (7216 fps) design goal in 1971, with a 3000-pound capsule.

NOTE:

POINTS A & B SHOW THE VELOCITY PERFORMANCE OF LEM AND TRANSTAGE, RESPECTIVELY, ADJUSTED TO THE SAME PLANETARY VEHICLE WEIGHT OF THE SOLID/MONO SYSTEM (19,500 POUNDS). PROPULSION MODULE WEIGHT IS ADJUSTED TO REFLECT PROPULSION SYSTEM IMPACT ON BUS, ADAPTER AND SHROUD WEIGHTS.

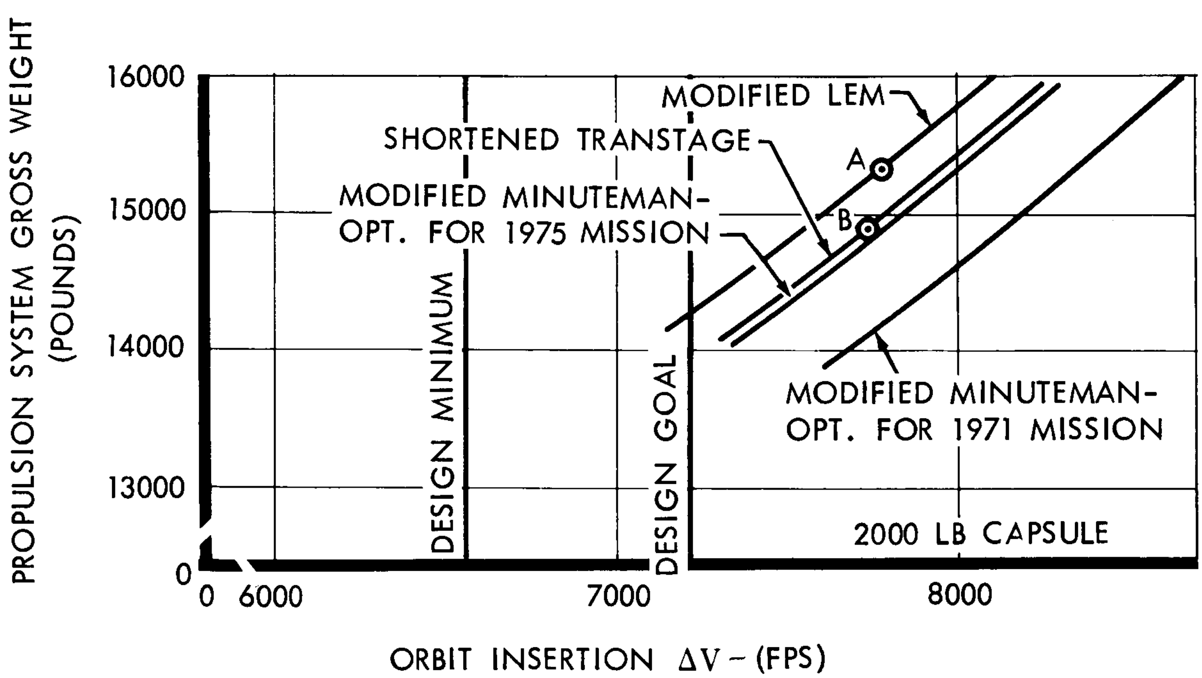


Figure 3.1-4: Orbit-Insertion Velocity Increment 1971 and 1973

NOTE:

POINTS A & B SHOW THE VELOCITY PERFORMANCE OF LEM AND TRANSTAGE, RESPECTIVELY, ADJUSTED TO THE SAME PLANETARY VEHICLE WEIGHT OF THE SOLID/MONO SYSTEM (20,500 POUNDS). PROPULSION MODULE WEIGHT IS ADJUSTED TO REFLECT PROPULSION SYSTEM IMPACT ON BUS, ADAPTER AND SHROUD WEIGHTS.

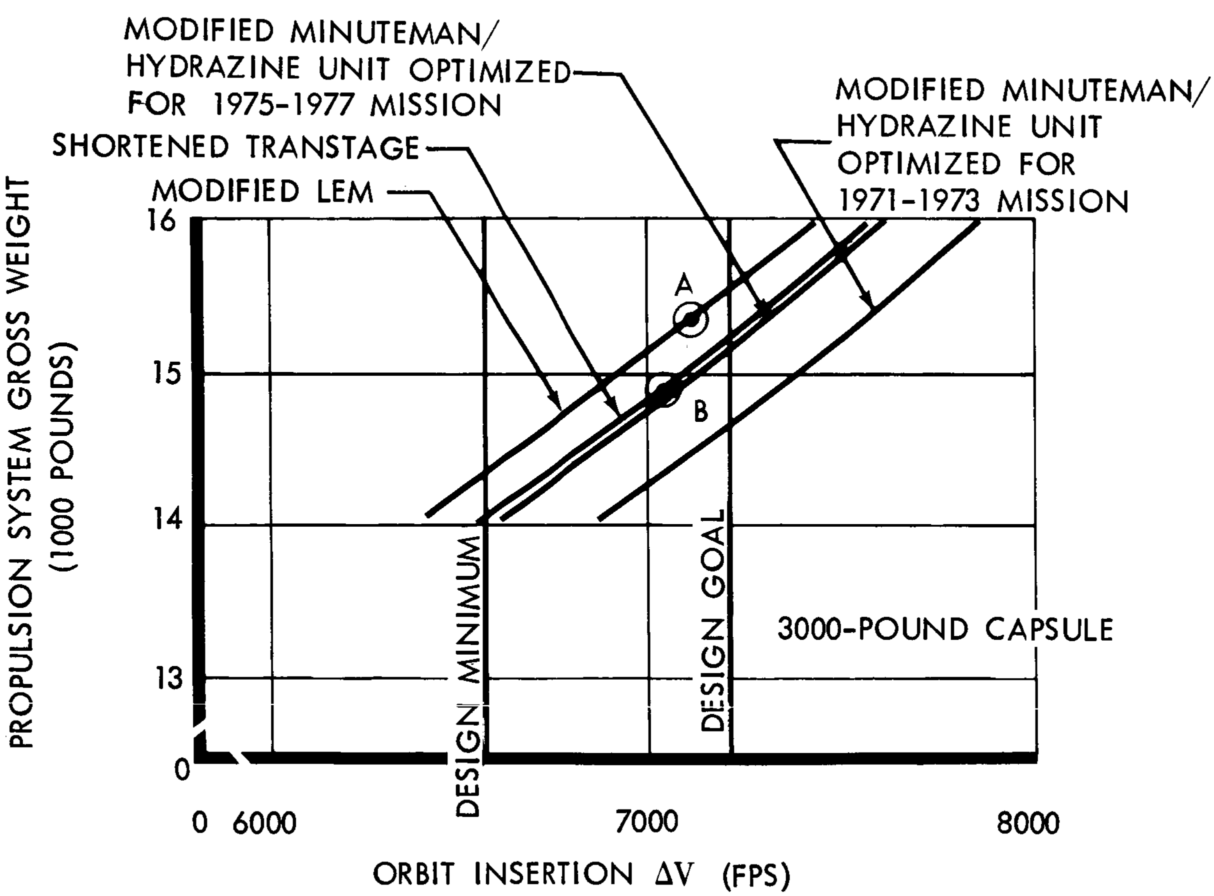


Figure 3.1-5: Orbit Insertion Velocity Increment 1971 and 1973

Table 3.1-5: CONFIGURATION WEIGHT COMPARISON

Configuration	Solid/Liquid (Modified Minuteman)		Shortened Transtage	LEM
Description	'71-'73	'75-'77	'71-'73	'71-'73
Secondary Propulsion				
Inerts	646	772	220	240
Usable Propellant	2495	3190	105	144
Main Propulsion				
Inerts	1390	1355	2375	1923
Usable Propellant	9839	9045	11437	11090
Structure	337	337	433	1240
Thermal Control	118	118	198	105
Cabling & Power Conditioning	25	25	92	92
Contingency	150	158	140	166
Total	15,000	15,000	15,000	15,000
Bus Weight Increment	0	0	+60	-100
Shroud Weight Increment	0	0	+90	-80
Adapter Weight Increment	0	0	0	-170
Total Weight Increment	0	0	+150	-350

'71-'73 Capsule Weight Used = 3,000 lb
'75-'77 Capsule Weight Used = 10,000 lb

* Boeing configuration designation

3.1.2.2 Velocity Increment Maneuver Accuracy

The variability in total impulse was evaluated for each of the propulsion concepts. The resulting velocity increment accuracy for trajectory correction maneuvers is as follows:

SYSTEM	VELOCITY INCREMENT ERROR SOURCE		
	ENGINE	ACCELEROMETER RESOLUTION	ACCELEROMETER NULL
Solid/Liquid	0.0016 fps	0.0377 fps	0.0025 ΔV fps
LEM Descent	0.168 fps	0.0377 fps	0.0025 ΔV fps
Transtage	1.85 fps	0.0377 fps	0.0025 ΔV fps

The velocity increment maneuver accuracy is particularly significant for orbit trim and midcourse correction maneuvers. Figure 3.1-6 shows the sensitivity of orbit period to periapsis velocity for various combinations of orbit periods and periapsis altitudes. A typical orbit with a 20-hour orbit period and a 1000 km periapsis altitude results in an orbit period sensitivity coefficient of approximately 6 minutes/meter/seconds. Adjustment of the orbit period is usually accomplished by an orbit trim maneuver at periapsis. If pointing errors are ignored, then, for a typical 50 meter/sec. trim maneuver, the solid/liquid concepts can adjust the orbit period to within 50 seconds; the LEM descent propulsion system can adjust the orbit period to within 68 seconds; and this maneuver, as performed by transtage, would result in a final orbit period which differs from the desired one by more than 252 seconds.

It is concluded that modified Minuteman/hydrazine systems and LEM provide significantly higher velocity increment maneuver accuracy than transtage.

3.1.2.3 Propulsion System Versatility

To provide versatility in mission planning, the Voyager Planetary Vehicle must be able to achieve all realistic orbits whose optimum total impulse requirements are equal to or less than the maximum propulsion capability.

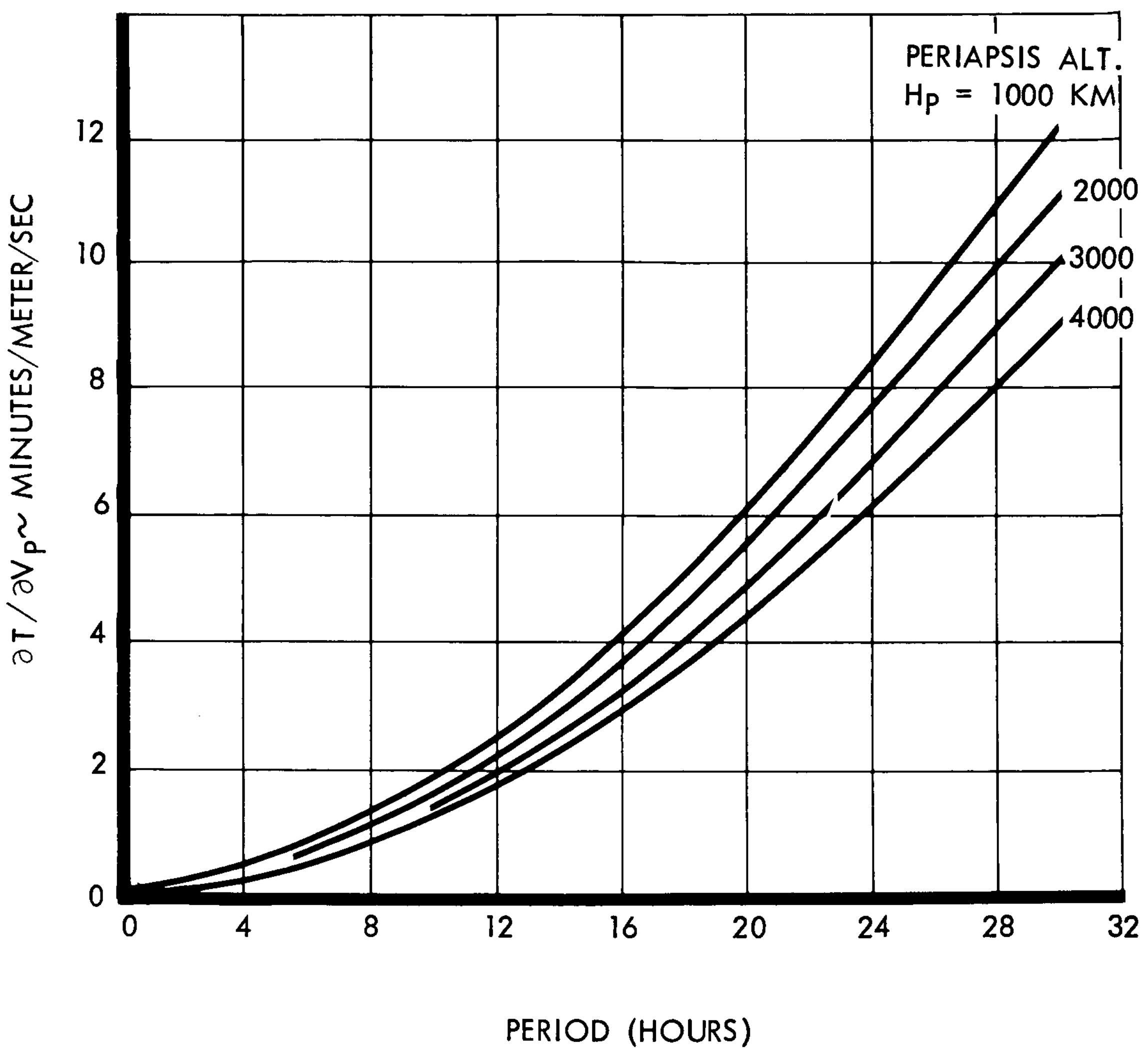


Figure 3.1-6: Orbital Period Sensitivity To Periapsis Velocity Errors

This is accomplished differently for propulsion systems with a fixed total impulse as compared to propulsion systems with an adjustable total impulse. In the following, thrust-terminated solids, propellant off-loading prior to mission launch, and the implications of excess liquid propellant in orbit are not considered. All liquid-propellant engines are considered as variable-impulse devices and all solid-propellant motors are considered as fixed-impulse devices as described in greater detail in Section 3.1 of D2-82709-6, Volume A, and in D2-82709-1. A variable-impulse engine can insert in a near-optimum fashion into a variety of orbits. Its versatility is limited only by the maximum ΔV that is available from the system. As an example, Figures 3.1-7(a) and 3.1-7(b) show the orbit-insertion performance of the LEM descent propulsion concept. For each orbit size, there is a Mars approach velocity (V_{HP}) for which a ΔV of 2.31 Km/sec is just sufficient to enter the given orbit in an optimum hyperbola-periapsis-to-ellipse-periapsis transfer. At lower V_{HP}, the extra impulse capability of the system allows some freedom in selecting orbit orientation.

For fixed-impulse orbit-insertion propulsion systems, a method is also available to insert into any desired orbit size and orientation, over a wide range of approach velocities. It requires a B-vector such that the approach hyperbola intersects the desired Mars orbit at a greater angle than that for the optimum-ΔV transfer. This increases the total impulse required to enter the orbit without changing the orientation or size of the final orbit. The principal limitation on the applicability of this technique occurs when the required B-vector is too low to satisfy planetary quarantine requirements. This limiting case occurs when low orbit periapsis altitudes are required in conjunction with low V_{HP}'s, and with

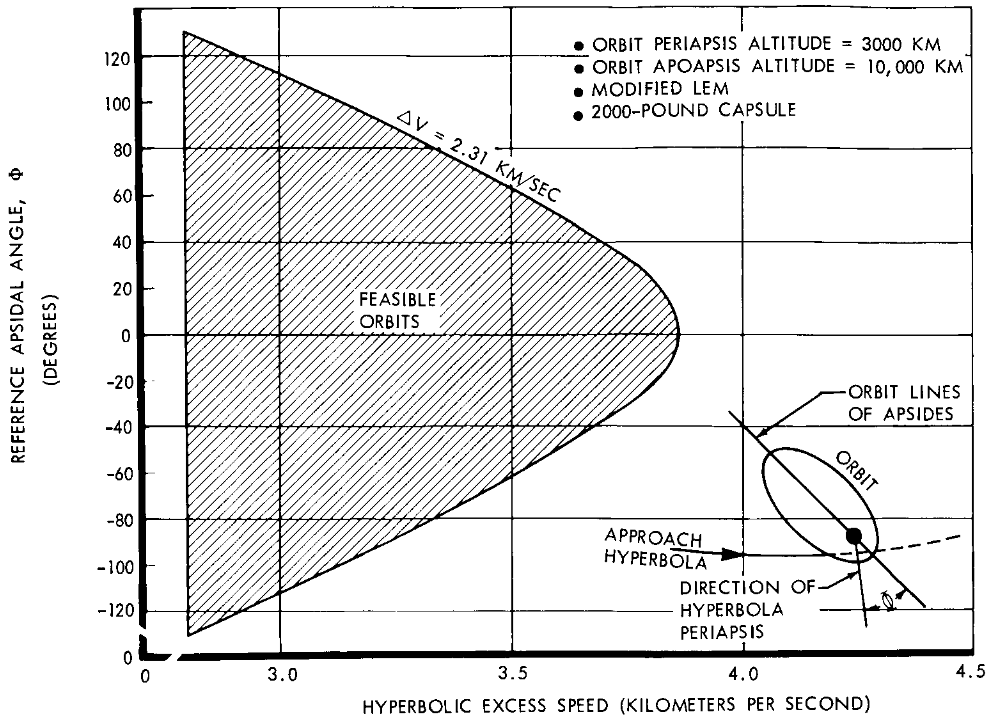


Figure 3.1-7a: LEM Orbit-Insertion Performance

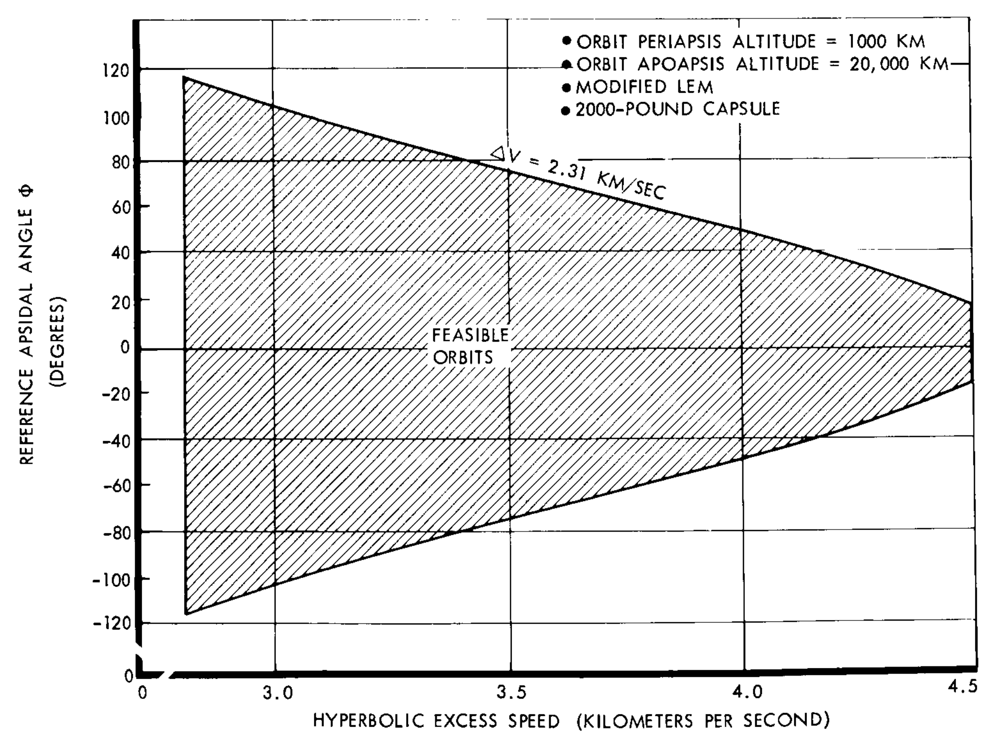


Figure 3.1-7b: LEM Orbit-Insertion Performance

orbit periapsis positions near the periapsis of the approach hyperbola (i.e., no apsidal rotation). A low periapsis prohibits lowering the B-vector. Also, orbits that require an optimum, i.e., low, ΔV for insertion are obviously more difficult to reach with fixed-impulse systems designed for a higher ΔV. Figures 3.1-8(a) and 3.1-8(b) show the performance of a solid-propellant orbit-insertion motor designed to orbit payloads at the highest practical ΔV for a 1971 mission. Figure 3.1-8(a) is similar to the performance chart for the LEM propulsion system, shown in Figure 3.1-7. The performance is limited at high V_{HP} by the maximum ΔV available from the motor. The low V_{HP} limit in Figure 3.1-8(b) results from inability to correct the B-vector sufficiently for combinations of low V_{HP}, low periapsis altitude, and the desired orientation of the orbit's line of apsides. This limit is easily removed by first entering an intermediate orbit with a higher periapsis altitude than the desired final orbit. An orbit vernier impulse is then applied at apoapsis to reduce the periapsis to the desired final altitude. Functionally, this is an orbit trim maneuver and can be accomplished simultaneously with the nominal periapsis trim maneuver.

Figure 3.1-9 presents the performance obtained from the modified Minuteman/monopropellant system, sized to the 1971 and 1973 missions, when vernier capability is provided by enlarging the midcourse and orbit-trim subsystem. (The solid motor is consequently reduced in weight to comply with propulsion system weight allocation.) Operating with a 2000-pound capsule, when 75 meters/sec is allowed for orbit insertion vernier, the high V_{HP} limit is moved slightly to the left, but the low V_{HP} limit is reduced considerably. With no more than 150 meters/sec orbit insertion vernier capability, the high V_{HP} limit is still above that for the competing

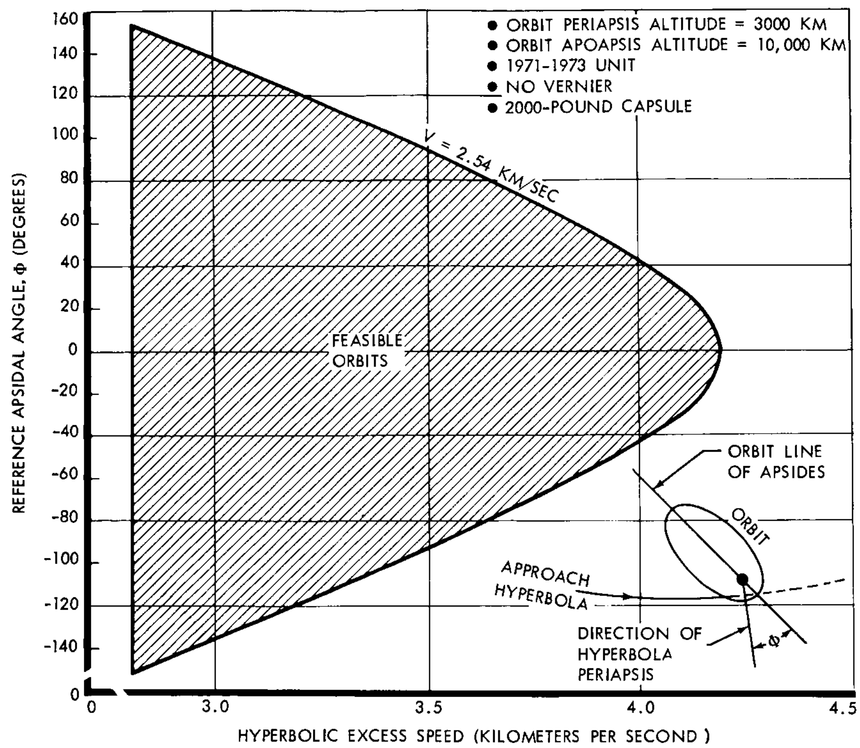


Figure 3.1-8a: Solid-Motor Orbit-Insertion Performance

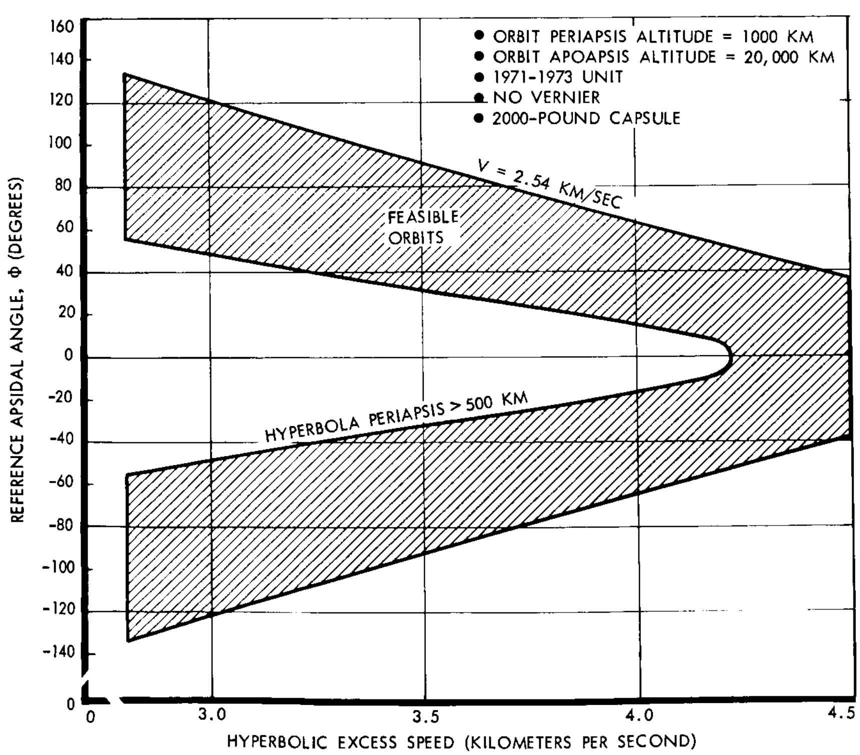


Figure 3.1-8b: Solid-Motor Orbit-Insertion Performance

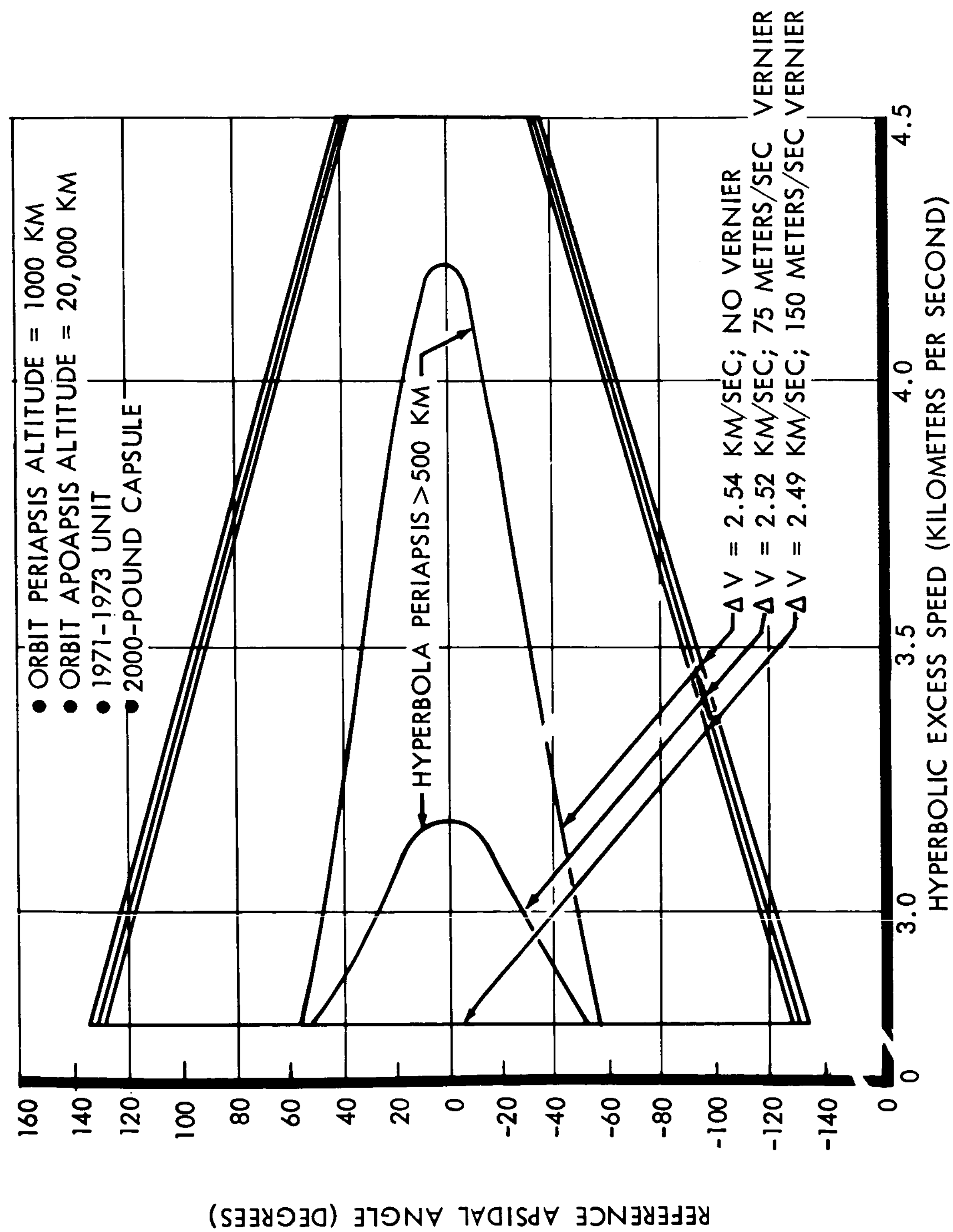


Figure 3.1-9: Solid/Liquid Orbit-Insertion Performance (Including Vernier)

bipropellant systems. The lower V_{HP} limit is completely removed. With a 3000-pound capsule, the solid motor results in a lower orbit insertion velocity increment than with a 2000-pound capsule. Consequently, the orbit insertion vernier requirement is only 100 meters/sec. The hydrazine subsystem is designed to accommodate a 3000-pound capsule. The additional 50 meter/sec vernier velocity requirement for a 2000-pound capsule is available from the hydrazine subsystem designed for a 3000-pound capsule without resizing. This is because the 2000-pound capsule requires less monopropellant for midcourse correction and orbit trim than does the 3000-pound capsule.

With orbit insertion vernier capability, the modified Minuteman/monopropellant propulsion system, sized for the 1971 and 1973 missions, attains all the orbits that the modified LEM and transtage systems can obtain. The operational complexity of the solid/liquid system is not significantly different from an "all-liquid" system, as indicated in Table 3.1-6.

Table 3.1-6: PROPULSION FLIGHT SEQUENCE COMPARISON

Mission Phase	LEM or Transtage	Modified Minuteman/ Monopropellant
Final approach aiming point	Can be selected for optimum insertion ΔV	Selected lower than the aim point for optimum insertion, but high enough to meet planetary quarantine requirements
Orientation for orbit insertion	Thrust vector oriented as required	Thrust vector oriented as required
Insertion maneuver	Thrust terminated at appropriate time	Total impulse fixed but known
Insertion errors	Determined by pointing and timing errors	Minimum of twice the sensitivities as LEM at insertion
Orbit trim	0 to 100 m/sec	0 to 250 m/sec (with vernier)

The 1971 mission orbit attainment versatility of the modified Minuteman/monopropellant unit, sized for the 1975 and 1977 missions, is shown in Figure 3.1-10. This unit includes a larger monopropellant subsystem than the one for the unit sized for the 1971 and 1973 missions. This is because midcourse corrections and orbit trim functions are provided to a heavier planetary vehicle in 1975 and 1977. Consequently no special provisions are required to provide orbit insertion vernier capability to this unit in 1971 and 1973. In fact, orbit insertion vernier is the only means by which the solid/liquid unit, sized for the 1975 and 1977 missions, accomplishes total orbit insertion in 1971. The solid motor inserts the planetary vehicle into an intermediate orbit. The hydrazine subsystem then augments the orbit insertion maneuver and verniers the planetary vehicle into the desired final orbit.

It is concluded that the solid/liquid unit, sized for the 1975 and 1977 missions, is as versatile as either LEM or transtage. The solid/liquid unit, sized for the 1971 and 1973 missions, is made as versatile as the other three competing systems by adding a modest orbit insertion vernier capability.

3.1.2.4 Minimum Impulse Bit Capability

Midcourse correction and orbit trim maneuvers may require extremely small velocity increments. The minimum impulse bit of a propulsion system depends on the capabilities of both the engine and the guidance and control subsystem. At engine shutdown, vehicle rates must not exceed gyro and reaction-control authority limits. Total impulse delivered by the four competing propulsion systems, at low total impulse levels, is as shown in Figure 3.1-11 as a function of total impulse required.

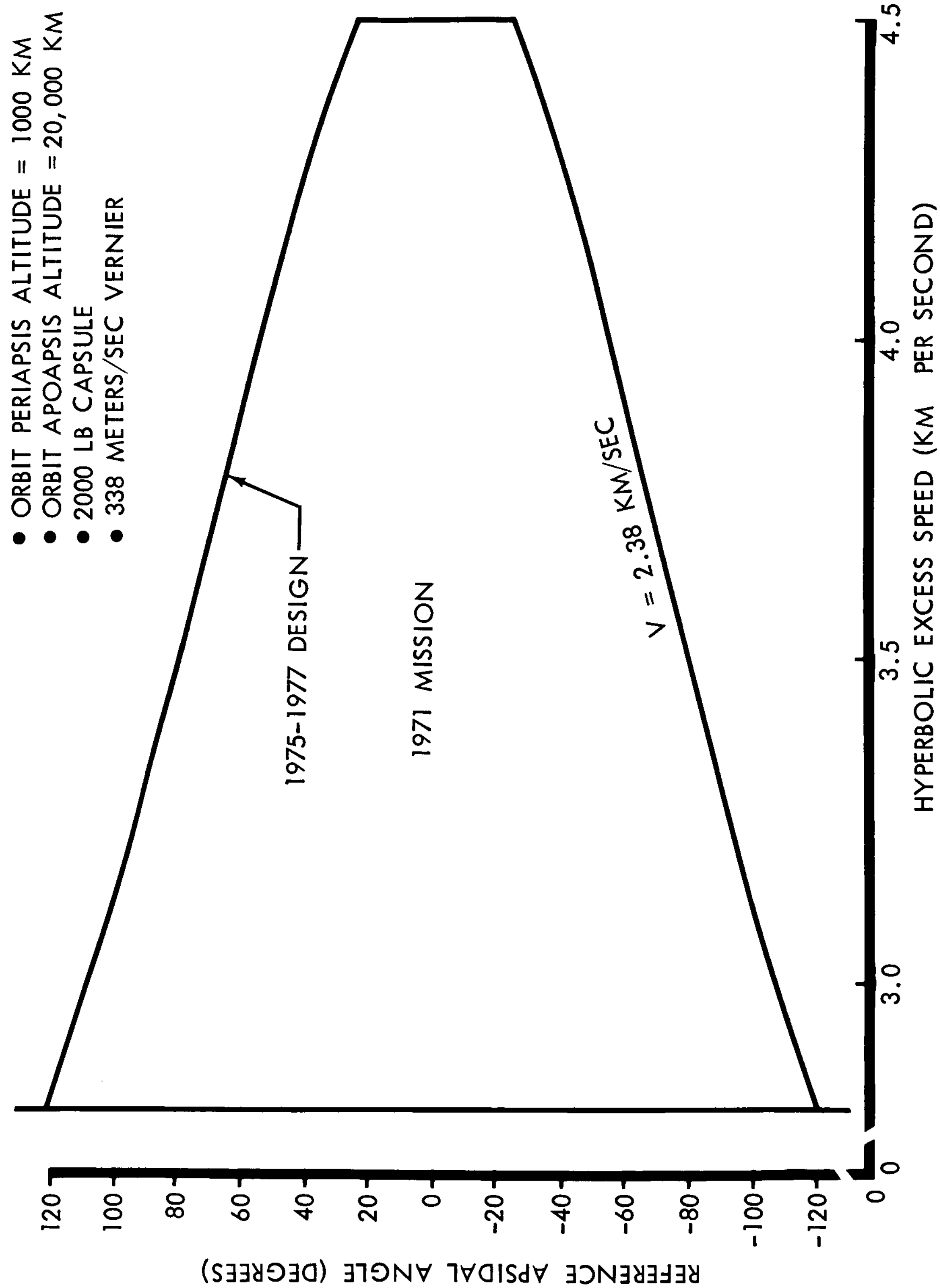


Figure 3.1-10: Orbit Insertion Versatility of the 1975-1977 Modified-Minuteman/Hydrazine Unit

PROPULSION MODULE	PROPULSION MINIMUM IMPULSE BIT (LB-SEC)	‡ EQUIVALENT MINIMUM ΔV METERS/SEC
TRANSTAGE (SETTLING)	50	0.0239
LEM (SETTLING AND TVC)	8*	0.00384
SOLID/LIQUID (LIQUID)	200	0.0957

* DESIGNED FOR PULSE MODE OPERATION
‡ 20,500-POUND PLANETARY VEHICLE

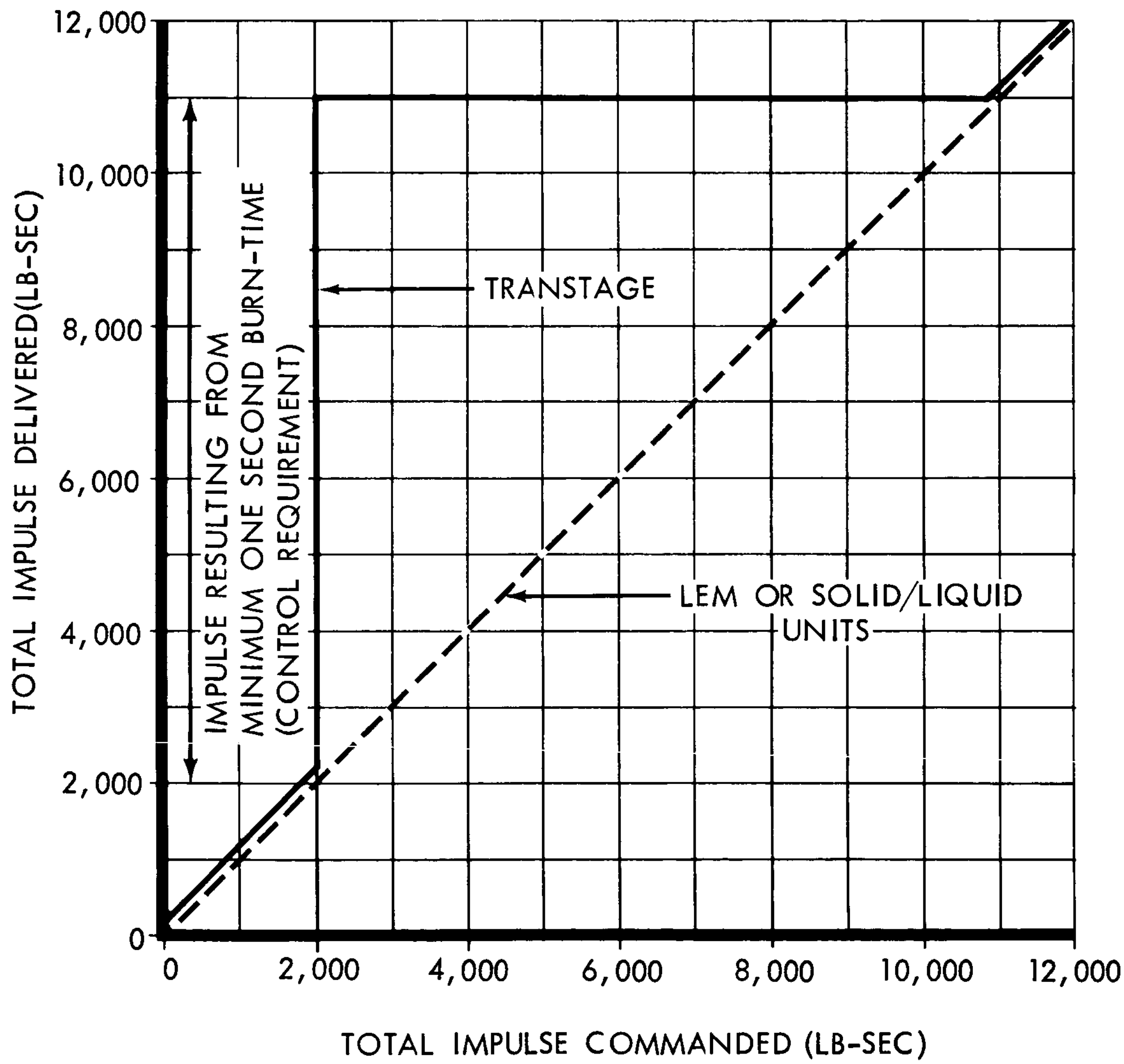


Figure 3.1-11: Midcourse Impulse-Bit Capability of Candidate Propulsion Systems

Both the LEM descent propulsion system and transtage use the main engine for midcourse correction and orbit trim. However, maneuvers with total impulse requirements less than the impulse required by the system for propellant settling will be terminated by the accelerometer prior to main engine firing. Consequently, the bipropellant stages have a lower minimum impulse bit capability than the solid/liquid units. This is because the total thrust level of the monopropellant settling system on LEM and transtage is lower than that of the midcourse and orbit trim monopropellant system of the solid/liquid units. On the other hand, maneuvers whose total impulse requirements exceed LEM and transtage settling-system total impulse, are performed as accurately by the solid/liquid units as LEM. Transtage exhibits poor low-impulse bit performance. This is because the minimum firing time for the main transtage engines is 1.0 second to provide acceptable vehicle rates at engine shutdown.

It is concluded that the modified Minuteman/hydrazine units and LEM have better overall performance at low total impulse levels than transtage.

3.1.3 Cost Savings

Candidate propulsion system costs accrue from design, developmental test and evaluation, test hardware, and flight hardware. For the 1971 mission, development, design verification, and type-approval testing contribute the greater part of system cost.

When considering missions through 1977, flight hardware accounts for the major part of system cost. Detailed, firm costs are not available at this time.

For the 1971 mission, when development and test costs dominate, cost differences between the four competing propulsion systems are expected to be small. When missions through 1977 are considered, where flight hardware costs dominate, the modified Minuteman/hydrazine unit is expected to result in least cost. This is because of the relatively low unit cost of the unmodified Minuteman motor.

3.1.4 Capability for Subsequent Missions

The orbit-insertion velocity increment capability of the candidate propulsion systems in 1975 to 1977 are shown in Figure 3.1-12. All systems have comparable orbit insertion ΔV capability.

For 1975 and 1977 missions, the minimum V_{HP} can be lower than that for the 1971 mission. However, the ΔV available from the propulsion systems for orbit insertion is much lower than in 1971 because of larger midcourse correction and orbit trim total impulse requirements. Consequently, the solid/liquid units provide as much versatility in orbit attainment in 1975 and 1977 as the two bipropellant stages without special orbit-insertion vernier allocations.

Figure 3.1-13 compares feasible orbits of the four propulsion systems for typical 1975 and 1977 missions. All orbits to the right of the performance line are feasible. Because the ΔV's from all three systems, with a 10,000-pound capsule, are nearly equal, the differences in orbit attainment capability for these three alternatives are not so large as they are in 1971 missions.

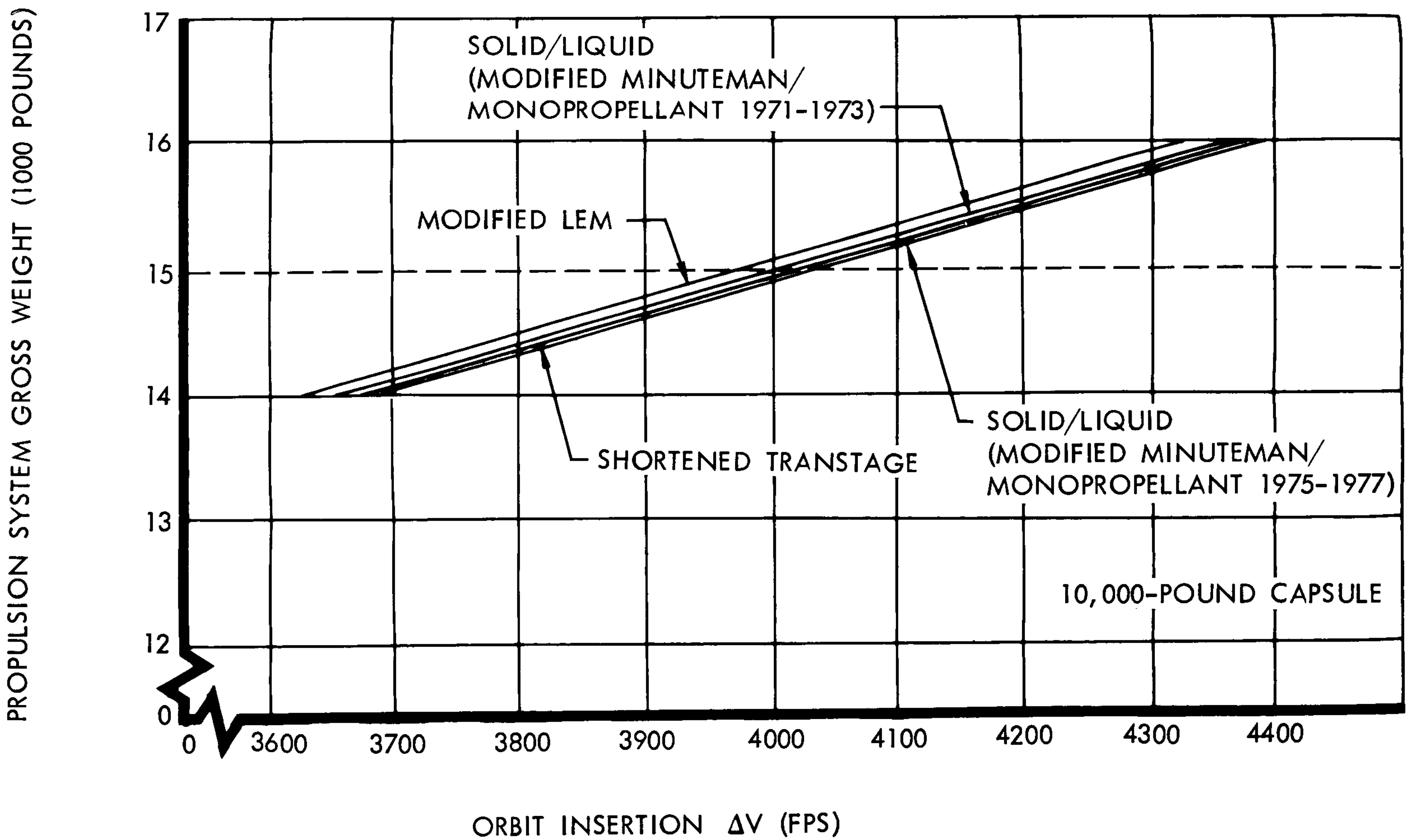


Figure 3.1-12: Orbit Insertion Velocity Increment — 1975 And 1977

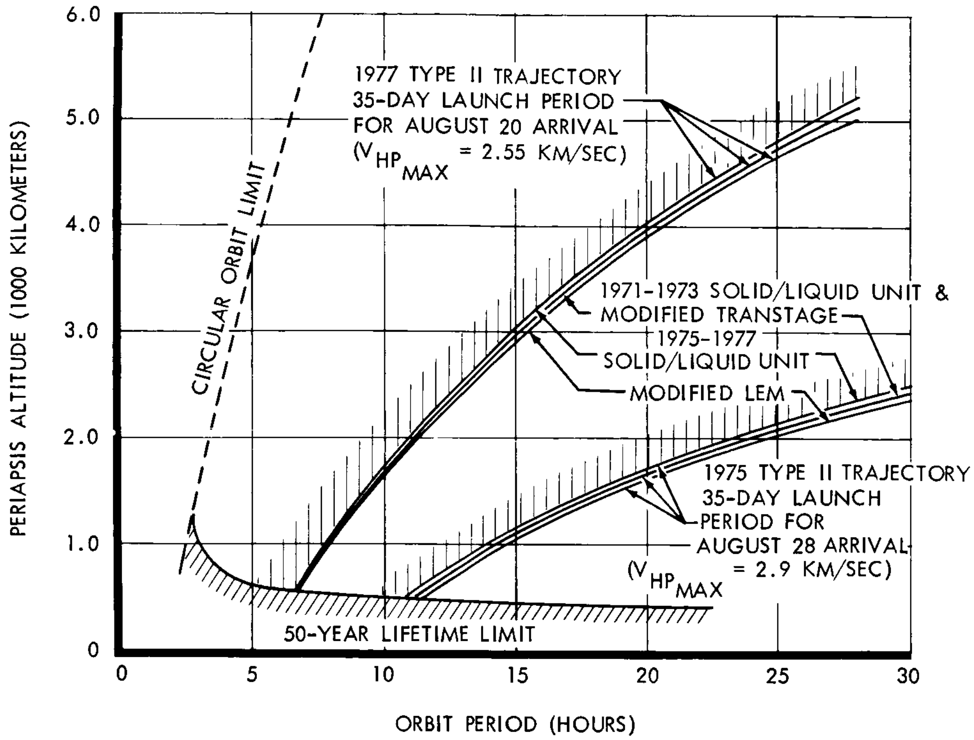


Figure 3.1-13: Orbits Available in 1975 And 1977 For Candidate Propulsion Systems

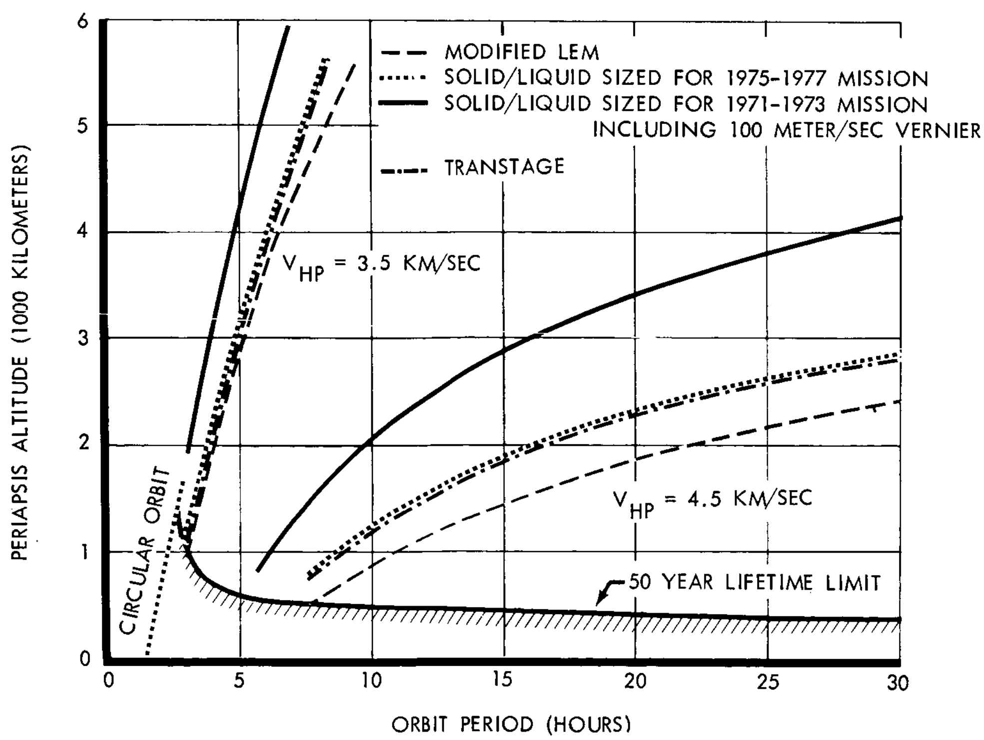


Figure 3.1-14: Excess 1971 Capability For Mars Orbit Insertion

It is concluded that the modified Minuteman/monopropellant unit, sized for the 1975 and 1977 missions, has the highest capability for subsequent Mars missions.

3.1.5 Additional 1971 Capability

The four competing propulsion systems all exhibit excess 1971 capability with a 2000-pound capsule. The modified Minuteman/hydrazine unit, sized for the 1971 and 1973 missions, is the only propulsion system with excess 1971 orbit insertion velocity increment capability for a 3000-pound capsule. Implications of the excess capability are shown in Figure 3.1-14. Periapsis-altitude/orbit-period combinations that are feasible from planetary quarantine constraints and propulsion capability are indicated for the range of allowable hyperbolic excess speeds. Excess orbit insertion capability is significant primarily at the higher hyperbolic excess speeds. This excess capability allows for achieving more circular, i.e., lower period, orbits. The maximum considered hyperbolic excess speed at Mars arrival is 4.5 Km/sec. The excess capability of the modified Minuteman/monopropellant units, transtage, and the LEM descent propulsion system, at this maximum hyperbolic excess speed is indicated on Figure 3.1-14.

It is concluded that the modified Minuteman/hydrazine unit, sized for the 1971 and 1973 missions, has the most additional capability in 1971. The modified Minuteman/hydrazine unit, sized for the 1975 and 1977 missions, has the second highest excess 1971 capability for a 2000-pound capsule. This capability is only slightly lower than that of the 1971 and 1973 unit. The LEM descent propulsion system has the least additional 1971 capability.

3.2 SELECTION RATIONALE

The competing characteristics discussion of the preceding section indicates that none of the four competing propulsion systems is significantly superior. The modified Minuteman solid/hydrazine subsystem units are, however, slightly superior to the LEM descent system and transtage in many competing characteristics. The modified Minuteman solid/hydrazine designs: 1) are conservative, 2) involve minimum technical risk, 3) make maximum use of existing hardware and available technology, and 4) rely on the demonstrated high reliability of the Mariner hydrazine subsystem and Minuteman motor.

In choosing between the two solid/hydrazine units, the unit sized for the 1975 and 1977 missions is preferable to that sized for 1971 and 1973, when all Voyager Mars missions through 1977 are considered. The unit sized for 1975 and 1977 offers 1) a single unit for all Voyager Mars missions without resizing; and 2) lower total cost. The modified Minuteman solid/hydrazine subsystem unit, sized for the 1975 and 1977 missions, is therefore selected as the preferred propulsion system for Voyager Mars missions.

In addition to the above, the candidate systems were compared on a point-rating basis similar to that used in Volume A, Section 3.11.3.2. A total of 1000 points was allocated among the competing characterisitcs, as shown in Table 3.2-1. The four competing propulsion designs were rated according to their ability to fulfill the competing characteristics for the Voyager Mars mission. The system that was best able to fulfill a competing characteristic was given a rating of 4. It was allocated the maximum number of points for that competing characteristic as given in

Table 3.2-1. The second best system was rated 3, and allocated three-fourths of the maximum allowable points, and so on. The results are summarized in Table 3.2-2.

Table 3.2-1: MAXIMUM POINT ALLOCATIONS FOR COMPETING CHARACTERISTICS OF CANDIDATE PROPULSION CONCEPTS

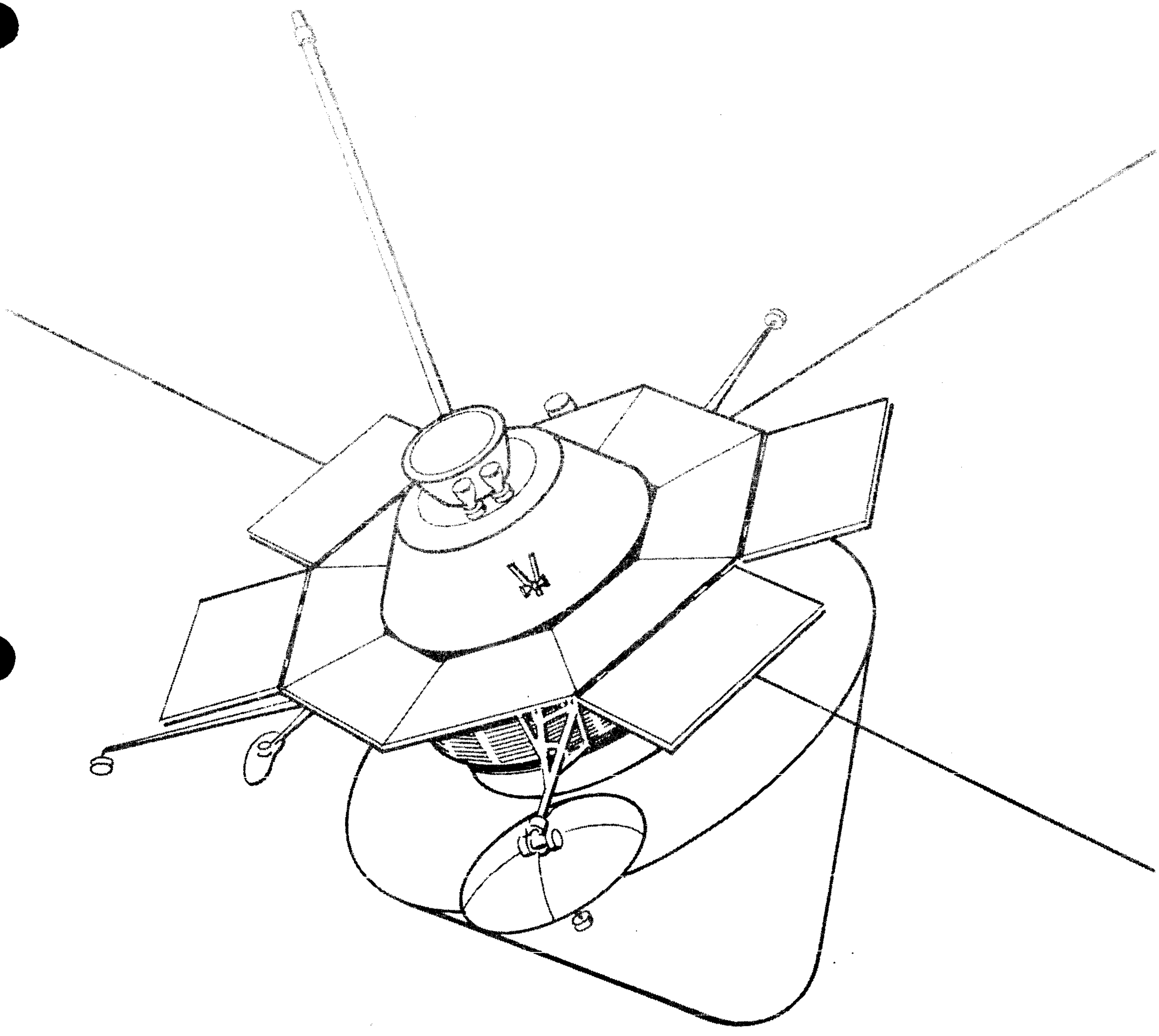
Probability of Mission Success		417
Extent of Modification	120	
Predicted Reliability	150	
Compatibility with Environment	80	
Compatibility with Planetary & Space Vehicles	60	
Compatibility with Planetary Quarantine	57	
Performance of Mission Objectives		236
Velocity Performance	120	
Propulsion System Versatility	80	
Velocity Maneuver Accuracy & minimum Impulse Bit	36	
Cost Savings		180
Capability for Subsequent Missions		111
Additional 1971 Capability		56
		1000

This process substantiates the selection of the modified Minuteman solid/hydrazine subsystem unit, sized for the 1975 and 1977 missions, as the preferred design. The selection process indicates that the solid/liquid units are superior to both transtage and the LEM descent propulsion systems. The selection process also reveals that the preferred solid/liquid unit, sized for the 1975 and 1977 missions, is as suitable for the 1971 mission as the unit sized specifically for the 1971 mission. The solid liquid unit sized for 1975/1977 missions results in lower cost and higher probability of mission success for subsequent missions because no resizing and requalification is required. This indicates a clear choice of the modified Minuteman-solid/hydrazine-subsystem unit, sized for the 1975 and 1977 missions, as the preferred design.

Table 3.2-2: NUMERICAL RATING OF CANDIDATE VOYAGER PROPULSION SYSTEM

Competing Characteristics	Modified Minuteman/ Monopropellant Sized for 1971-1973	Modified Minuteman/ Monopropellant Sized for 1975-1977	LEM/Mono	Transtage/Mono
Extent of Modification	*(4)120	(4)120	(4)120	(3)90
Reliability	(4)100	(4)100	(2)50	(3)75
Compat. With Environment	(4)80	(4)80	(3)60	(3)60
Compat. With P/V & S/V	(4)60	(4)60	(3)45	(2)30
Compat. With Plan. Quarantine	(4)57	(4)57	(4)57	(4)57
PROBABILITY OF MISSION SUCCESS	417	417	332	312
Velocity Performance	(4)120	(3)90	(2)60	(3)90
Versatility	(3)60	(4)80	(4)80	(4)80
Velocity Maneuver Accuracy	(4)36	(4)36	(3)27	(2)18
PERF. OF MISSION OBJECTIVE	216	206	167	188
COST SAVINGS	(4)180	(4)180	(4)180	(4)180
CAPAB. FOR SUBSEQUENT MISSIONS	(3)83	(4)111	(2)56	(3)83
ADDITIONAL 1971 CAPABILITY	(4)56	(3)42	(2)28	(3)42
TOTAL	952	956	763	805

*Numerical Rating ((4) is highest rating)



PART II SUPPORTING TRADE STUDIES

4.0 SOLID/LIQUID SYSTEM OPTIMIZED FOR 1971 - 1973

4.1 SOLID-MOTOR SYSTEM

Three solid-motor design concepts were considered: (1) Modification of an existing motor; (2) A new motor; and (3) A motor cluster.

The following describes the evaluation of each of the concepts and the selection rationale for the preferred configuration.

4.1.1 Modification of an Existing Motor

4.1.1.1 Motor Selection

Candidates--The following three production motors were considered: Polaris A-III second stage, Minuteman Wing II second stage, and Minuteman Wing VI second stage. These were considered because their weight and total impulse were sufficiently close to that required.

Competing Characteristics--The primary existing motor competing characteristics are: demonstrated reliability, high velocity performance, and propellant loading.

Selection Rationale and Discussion--The Polaris A-III and Minuteman Wing II motors were rejected for the following reasons:

1) Polaris A-III--The nozzle expansion ratio is only 8 to 1, which would be difficult to improve because of the four-nozzle configuration. Its Class 9 propellant is undesirable from handling and safety considerations.

2) Minuteman Wing II--The comparatively small case diameter results in a long configuration. Also it has four nozzles with a low expansion ratio (16 to 1).

The motor remaining for consideration is the Minuteman Wing VI, second stage. It has had 16 successful flights in 16 attempts. The single-nozzle expansion ratio is 24.8 to 1, resulting in a delivered vacuum-specific impulse of * . The propellant loading, * pounds, is in excess of that required for Voyager.

4.1.1.2 Modifications Required (Exclusive of TVC)

Propellant Loading--To meet the propulsion module weight allocation of 15,000 pounds, the propellant loading of the modified Minuteman motor is restricted to 9,839 pounds. This off-loading is accomplished most reliably by maintaining grain geometry and using a 30-inch shorter case and mandrel. The resulting chamber pressure and thrust time traces are shown in Figure 4.1-1. Induced vehicle g-loading during orbit insertion is shown in Figure 4.1-2. The vehicle weight penalty resulting from this g-loading is shown in Figure 4.1-3. The spacecraft is designed to withstand the capsule-off g-loading.

Nozzle Exit Cone Extension--Vehicle configuration trades reported in Volume A favor mounting of solar panels on the vehicle aft portion. As shown in Figure 4.1-4, exhaust plume radiation of the existing motor causes excessive heating of the solar panels. Extending the nozzle exit cone 15 inches to an area ratio of 32.5 solves this problem, and provides an adequate design margin. Changing the exit cone this amount is considered a straightforward modification.

*See D2-82709-10 Classified Supplement - Reference Page 22

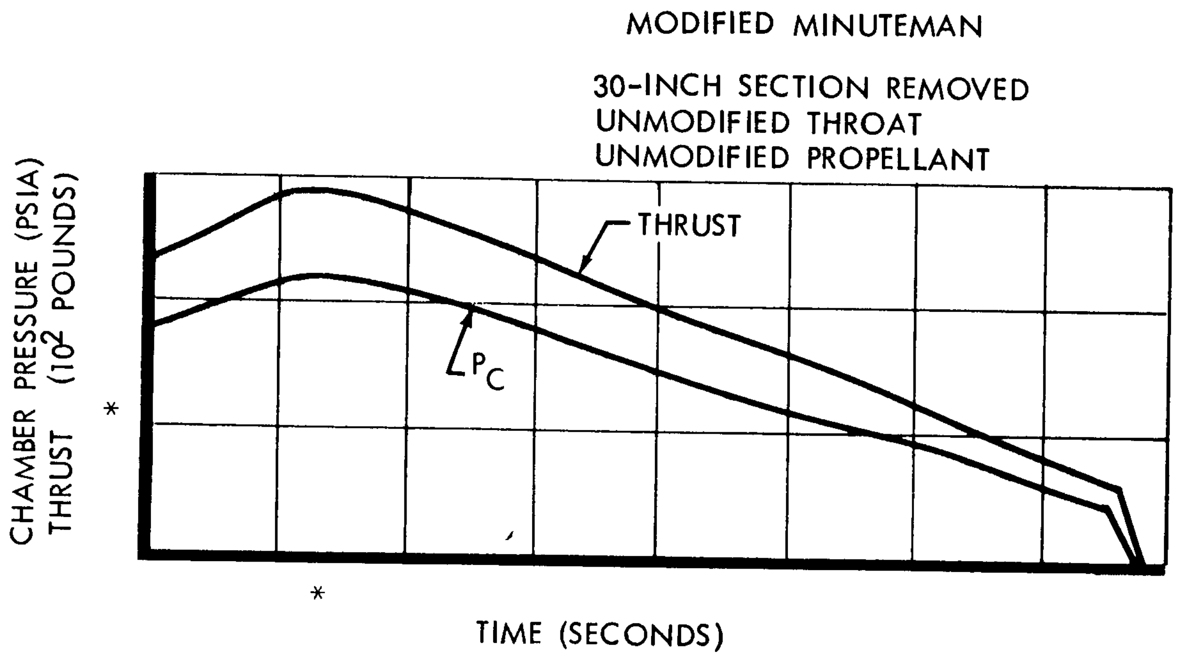


Figure 4.1-1: Chamber Pressure And Thrust Time Traces

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FOR CLASSIFIED INFORMATION

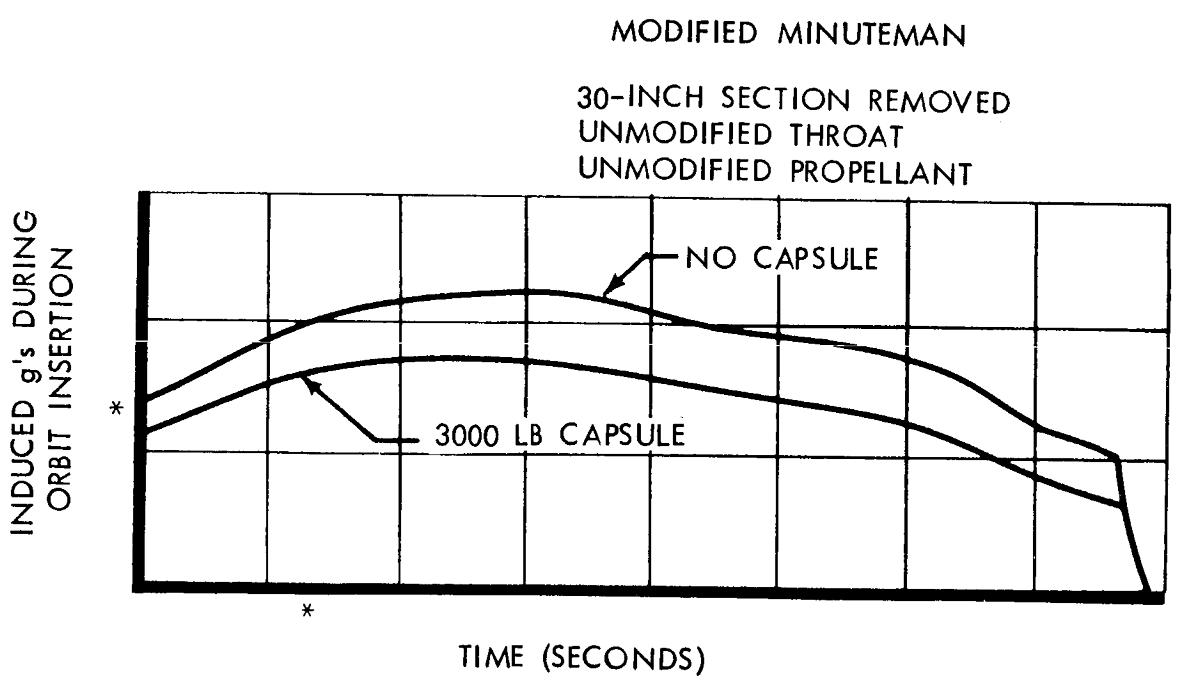


Figure 4.1-2: Orbit Insertion g Loading

*See D2-82709-10 Classified Supplement - Reference Page 23

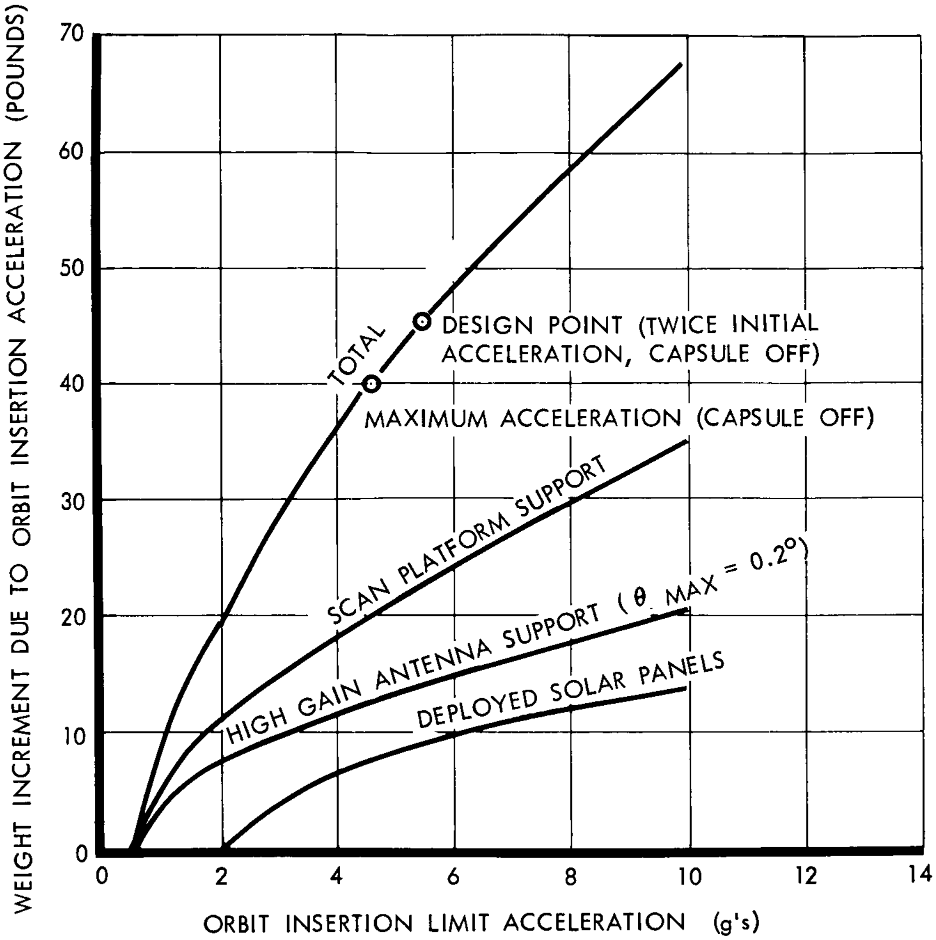


Figure 4.1-3: Effect Of Orbit Insertion Acceleration On Spacecraft Weight

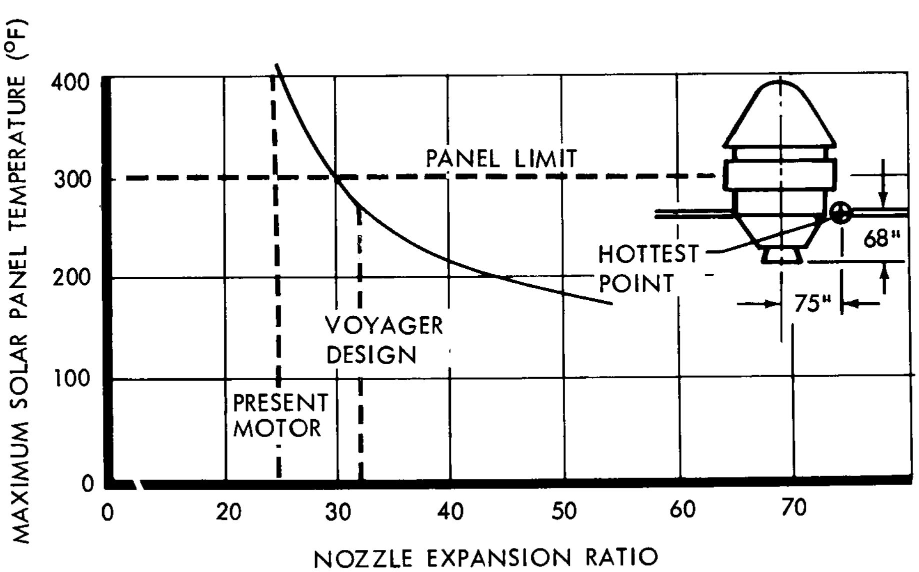


Figure 4.1-4: Solid Motor Exhaust Plume Heating — Modified Minuteman

4.1.2 New Motor

A preliminary motor specification was sent to propulsion vendors. Three of the submitted designs are summarized in Table 4.1-1. The performance spread is indicative of what can be achieved with a new design. The modified Minuteman motor is added for comparative purposes.

Table 4.1-1: MOTOR SUMMARY

PARAMETER	COMPANY A	COMPANY B	COMPANY C	MODIFIED MINUTEMAN
Diameter, in.	54	54	52	52
Length, in.	116	140	145	150.3
Nozzle Expansion	70	85	60	32.5
Specific Impulse, sec	*	302.6	*	*
Type Case	Fiberglas	Fiberglas	Titan.	Titan.
Mass Fraction without TVC	0.90	0.926	0.91	0.904

4.1.3 Clustered Motors

A clustered motor configuration was considered for improving orbit insertion probability of success. The following rules were observed in designing the cluster.

1) All motors in the cluster must fire to achieve the desired orbit.
2) A practical orbit must be achieved even if one motor does not fire.
3) Orbit insertion must be accomplished without a capsule (TVC-c.g. considerations).

Figure 4.1-5 shows the evaluated configuration. Seven motors were selected as a compromise between minimizing performance loss during motor-out operation and packaging problems. With the 5-degree effectiveness of

*See D2-82709-10 Classified Supplement - Reference Page 24

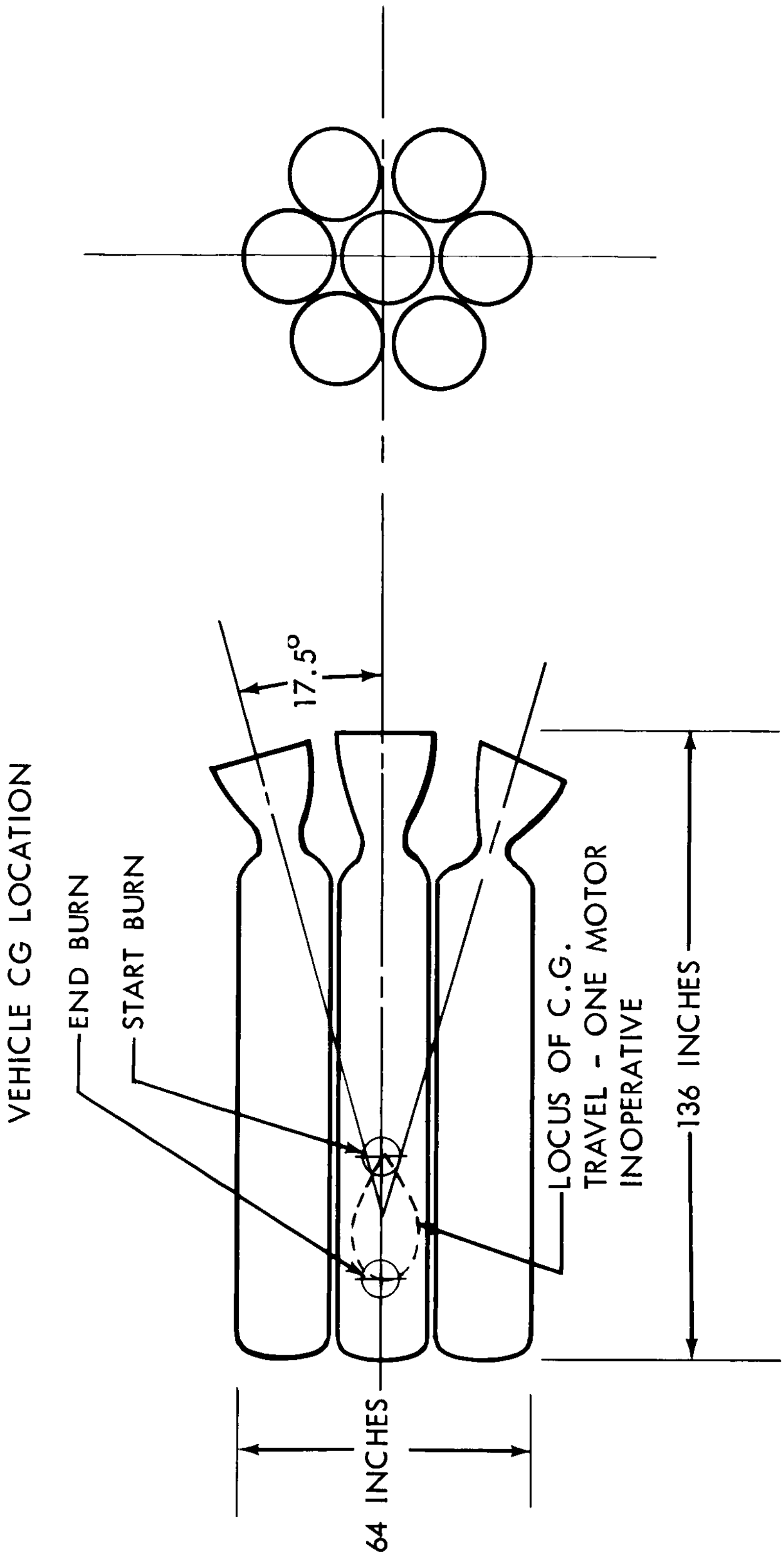


Figure 4.1-5: Clustered Motor Configuration

secondary fluid-injection TVC in each motor, a nozzle cant angle of 17.5 degrees is required to maintain thrust vector control during motor-out operation with capsule off. No existing motor could be found that is suitable for this application. Preliminary designs were requested from motor manufacturers. Their estimated mass-fractions were between 0.75 and 0.80, without TVC, because of the geometry and thrust level restrictions. This results in a total weight increase of 1200 pounds over an equivalent single-motor configuration.

4.1.4 Solid-Motor Selection

Candidates--The following concepts were considered: Modified Minuteman Wing VI second stage, a new motor, and a seven-motor cluster.

Competing Characteristics--The primary competing characteristics are:

1) Probability of mission success;
2) Performance of mission objectives;
3) Cost.

Selection Rationale and Discussion

New Motor--Figure 4.1-6 shows a performance comparison of the three concepts. A new solid motor design offers significant velocity performance gains over the modified Minuteman motor, and could demonstrate high reliability with sufficient testing. A decision in favor of a new solid motor cannot be made, however, without firm cost data to establish its cost effectiveness. Solid motor specifications are being released to qualified propulsion vendors. These specifications can be met by either a modified Minuteman motor or a new solid. The preferred solid motor design selection will

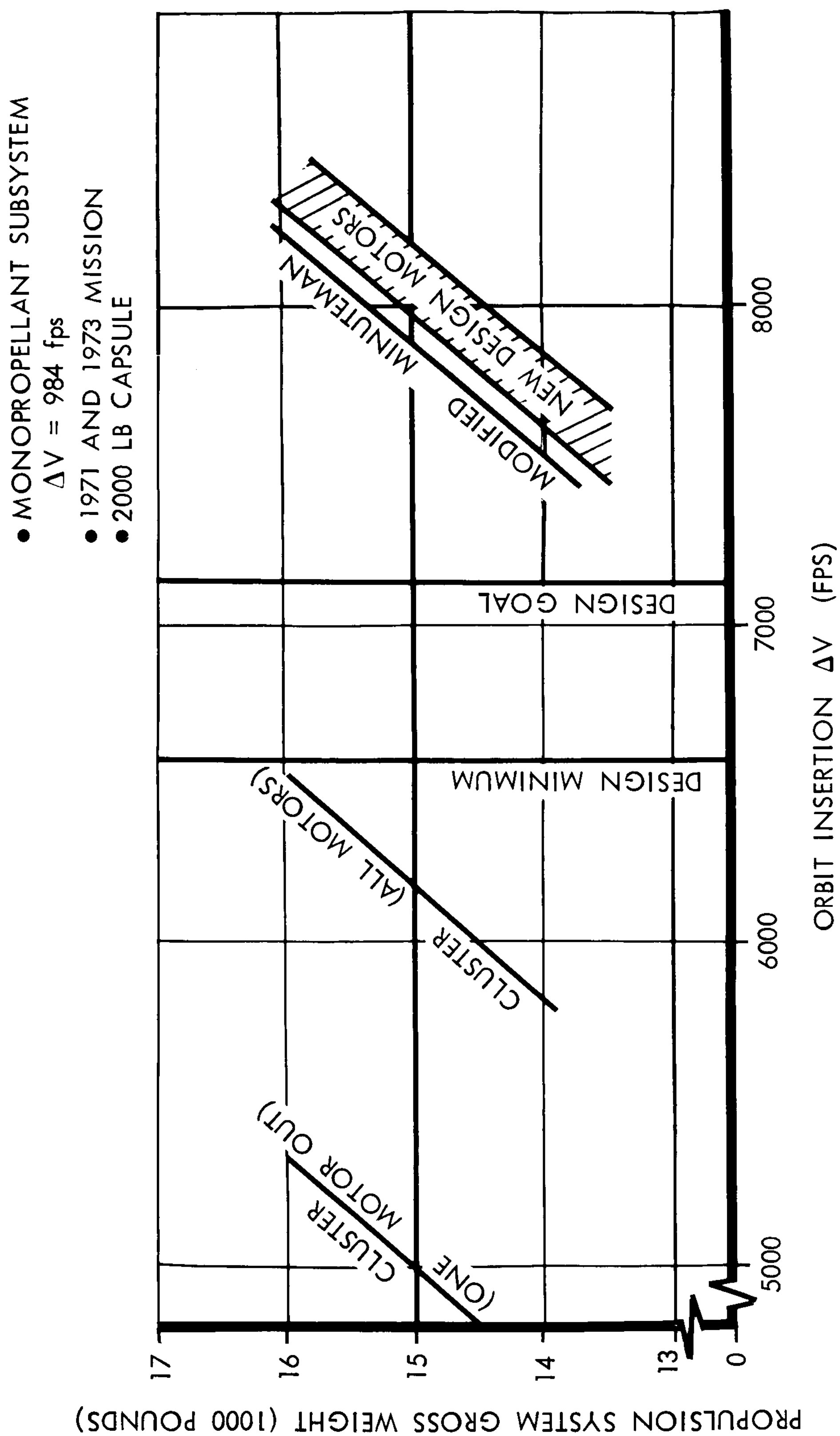


Figure 4.1-6: Performance Of Solid Motor Concepts

be reviewed with the aid of forthcoming propulsion vendor design, schedules and firm cost data.

Motor Cluster--This configuration is rejected as it cannot meet the minimum orbit insertion velocity increment requirement without exceeding weight allocations.

Modified Motor--The modified Minuteman second stage is the recommended solid motor design. It represents a minimum technical risk. A conscious effort was made to keep modifications to a minimum. Motor-induced swirl torques are well defined and small. Motor heat soakback after burn-out is compatible with vehicle requirements.

4.1.5 Thrust Vector Control

4.1.5.1 Liquid Secondary Injection

Description--An analysis was made to optimize and characterize the performance of liquid secondary injection TVC for Voyager. Liquid secondary injection offers high-response-rate capability for thrust vector control of solid-propellant motors. Disturbance torques can be held to a minimum. This is because, unlike liquid-propellant stages, tight tolerances on lateral c.g. and motor thrust alignment are feasible. The use of solid propellant eliminates fuel-slosh dynamic coupling into the TVC autopilot loop. Tail-wags-dog dynamic couplings, usually associated with gimbaled engines, are also absent.

With a 2000-pound capsule, the longitudinal distance from the c.g. of the preferred solid motor design to the effective thrust vector trunnion point ranges from approximately 88 inches (start burn) to 105 inches (end burn). The change in TVC autopilot loop gain (due to changes in longitudinal c.g. to trunnion distance and spacecraft moment-of-inertia) during motor burn presents no control problems. The 3σ tolerance on lateral c.g. offset can be held to 0.21 inches at start burn and to 0.26 inches at end burn.

These tolerances account for the deployment failure of appendages, and allow for uneven monopropellant consumption. A motor thrust angular misalignment of 0.25 degree was assumed. Thrust vector inertial pointing errors (one axis) can then be held within 0.3 degree for practical autopilot gains. This is well within the 1-degree pointing error budgeted to the TVC system.

Motor ignition-TVC transients are shown in Figure 4.1-7. A functional block diagram of the simulated autopilot loop is also shown. Ideal rate and position feedback were assumed for purposes of comparison between different systems. Structural coupling into the autopilot loop did not present any problem.

Secondary factors to be considered in future analyses include non-linearities of the pintle-valve hydraulic-actuator combination, particularly the negative spring effect on the pintle due to fluid flow. Fuel slosh in the monopropellant tanks should also be considered, although fuel slosh dynamic coupling problems are not anticipated.

It is concluded that liquid secondary injection TVC is compatible with the preferred Voyager design. The selection of the preferred liquid injection TVC system follows.

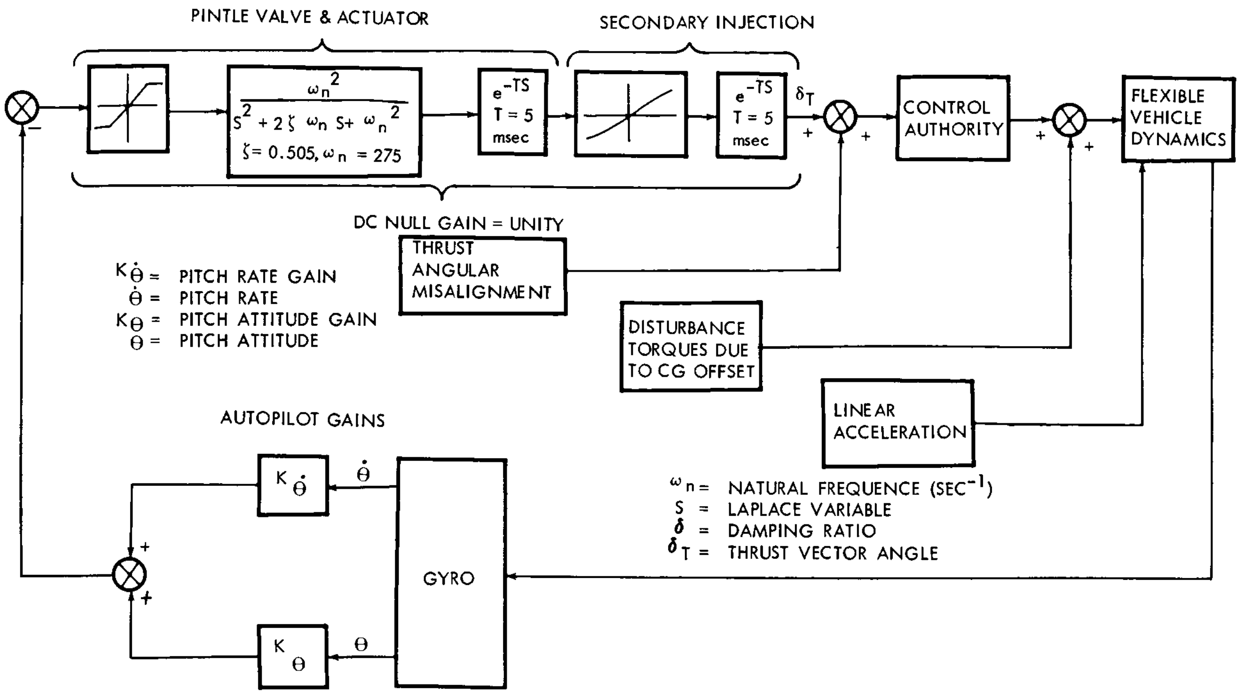


Figure 4.1-7a: Autopilot Block Diagram

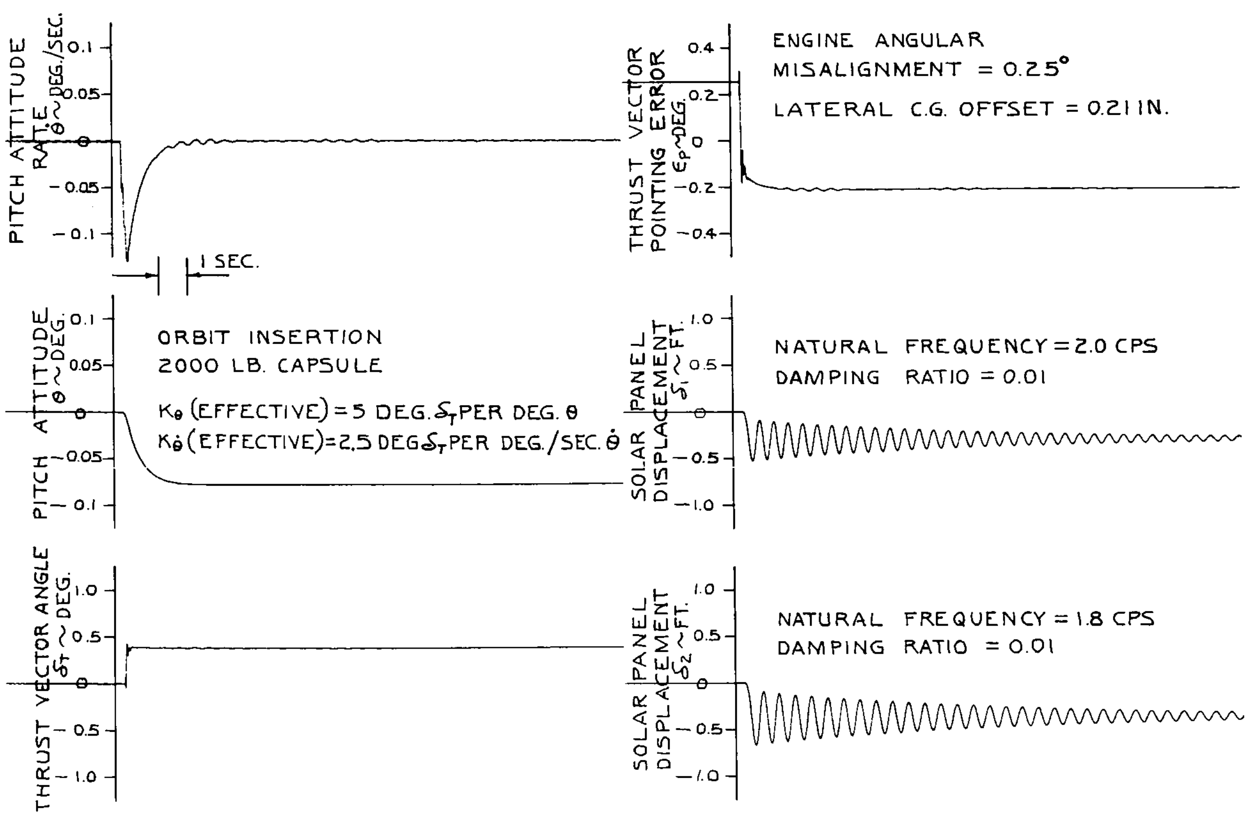


Figure 4.1-7b: Start Burn Transients

Figure 4.1-7: Secondary Injection TVC Mechanization And Dynamics

Candidates--The following candidates for the thrust vector control system were considered: the existing Minuteman TVC system, a modified Minuteman system, and a new system.

Competing Characteristics--The primary competing characteristics are reliability, use of proven components, and weight.

Selection Rationale and Discussion--The existing and new-design TVC systems were rejected for the following reasons:

1) Existing Minuteman System--The freon tank material is 17-7PH stainless. It is magnetic and therefore unacceptable. The pressurization system using solid-generated gases is unsatisfactory as it may cause solar panel and sensor contamination when dumping excess gas overboard. The power requirements of the electro-hydraulic pumps are excessive.
2) New System--A new system was designed that corrected the deficiencies of the existing Minuteman. It was rejected because of the development required. It did not offer significant weight and control dynamics advantages.

The preferred design is one using the existing Minuteman system with modifications. The hot-gas pressurant is replaced by a cold-gas (nitrogen) system. The toroidal tank material is changed to titanium, which is magnetically satisfactory. The electro-hydraulic system is replaced by stored, regulated nitrogen pressurant. The injector valve is modified to operate with Freon in the servo section instead of with the currently-used hydraulic oil. The last modification is simple. Valves of this type are now in use on the Polaris missile.

4.1.5.2 Roll Control

A reaction-control system must be provided to combat the roll disturbances during orbit insertion. These are caused by solid-motor vortex flow and transient thrust vector misalignment during orbit insertion. System authority is determined by motor-generated roll disturbances, which are more severe than those resulting from thrust vector misalignment. The magnitude of these roll torques were determined from Minuteman Wing VI flight-test data and are shown in Figure 4.1-8.

Candidates--The orbit insertion roll-control systems considered were:

1) The hot-gas roll-control system used in conjunction with the motor in its Minuteman application;
2) A nitrogen cold-gas system.

Competing Characteristics--The following characteristics were considered in the final selection of the roll-control system:

1) Reliability;
2) Compatibility with the spacecraft;
3) Weight.

Selection Rationale--The existing hot-gas roll-control system was rejected for the following reasons:

1) The solid-generated hot-gas exhaust products are unsatisfactory because of possible solar-panel and sensor contamination;
2) System capability is much greater than that required for Voyager application, and it results in weight penalties.

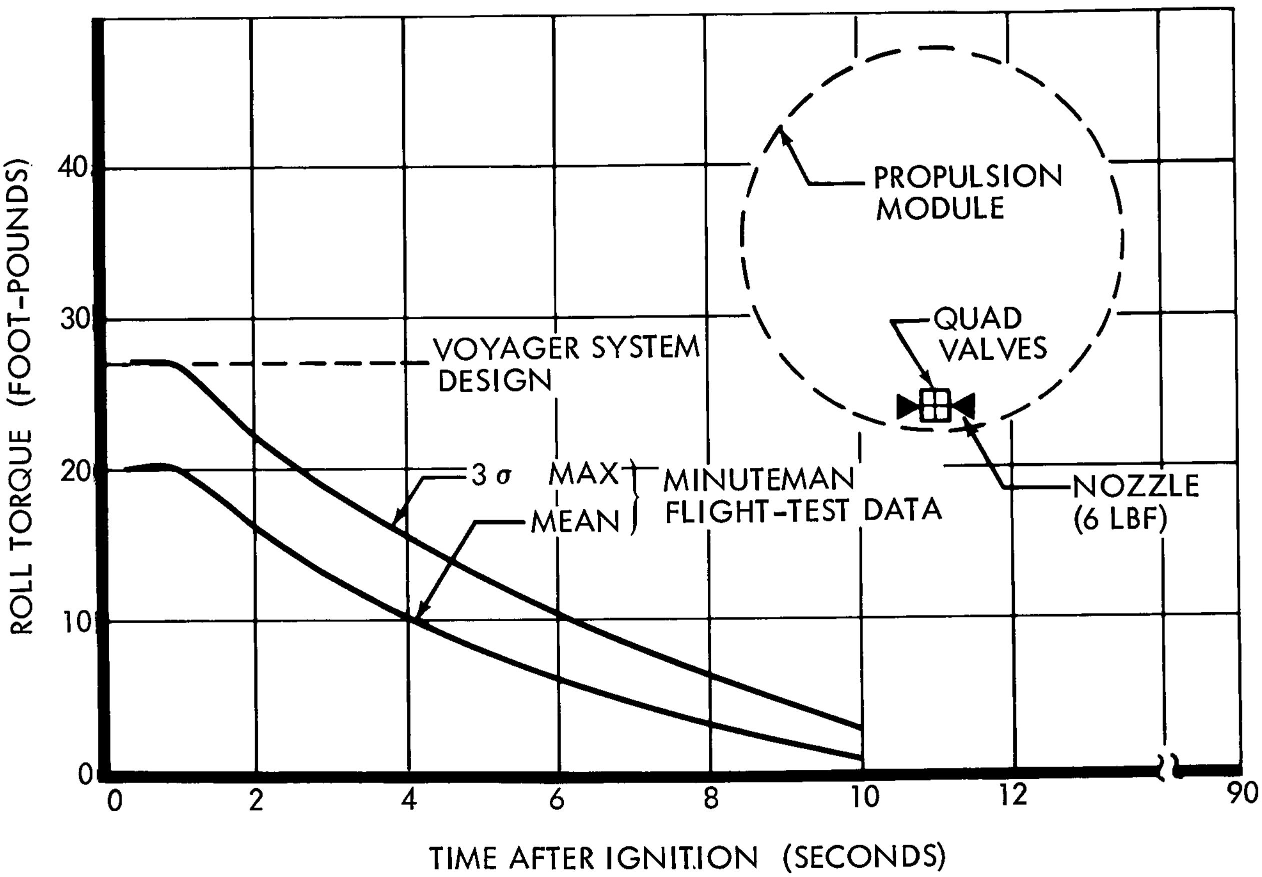


Figure 4.1-8: Roll-Control System Design Torque

The preferred design is a nitrogen cold-gas system using the TVC pressurant as the propellant. Two 6-pound thrusters are employed using quad-solenoid valves for redundancy.

4.2 LIQUID-PROPELLANT SYSTEMS

Both monopropellant and bipropellant engine systems were considered, with single and multiple engine installations. Monopropellant systems used a spontaneous catalyst for initiating and sustaining propellant decomposition. Bipropellant systems used hypergolic reaction for ignition and to sustain combustion. Thrust vector control techniques considered included jet vanes, gimbaled engines, and differential throttling. Both canted and noncanted engine installations were examined. Propellant storage, expulsion, and pressurization methods required for these systems were analyzed, as were valving and plumbing arrangements and thermal control requirements. The studies leading to the preferred liquid system are described below.

4.2.1 Monopropellant System

4.2.1.1 Monopropellants

Description--Screening of candidate monopropellants has been reported previously in Task A. Anhydrous hydrazine (N_2H_4), was selected as the most applicable monopropellant for Voyager on the basis of: accumulated space experience, stability and storage characteristics, reliability, and specific impulse. This selection remains unchanged.

The freezing point of hydrazine (34.5°F) may be depressed by the addition of water, if required. Hydrazine vacuum specific impulse (steady-state) is nominally 235 seconds, based on Ranger and Mariner experience. A specific impulse of 237 seconds is feasible at the following conditions:

thrust - 200 lb, chamber pressure = 150 psia, expansion ratio = 50. It is understood that a specific impulse of * percent seconds has already been demonstrated by JPL with a Mariner engine at a 50:1 nozzle expansion ratio.

4.2.1.2 Monopropellant Engines--Type

Description--The engine assembly consists of a thrust chamber (reactor), nozzle, engine valve, thrust vector control assembly, and associated support structure. The reactor contains a catalyst to promote hydrazine decomposition.

The radiation-cooled reactor and nozzle assembly, selected in Task A, is retained on the basis of demonstrated flight experience and relative insensitivity to operating duration. The reactor walls and nozzle are fabricated of Haynes 25 alloy. The injector is fabricated of aluminum.

The Shell 405 (Shell Development Company) spontaneous catalyst chosen in Task A is also retained. It is less complex and inherently more reliable than catalyst heating systems or ganged hypergolic slugs. This catalyst has now reached a sufficient development status to be considered ready for flight use. Within the propellant temperature boundaries of $+40^{\circ}$ to $+135^{\circ}F$, the 405 catalyst provides consistent propellant ignition. It sustains propellant decomposition for time periods in excess of Voyager requirements.

4.2.1.3 Thrust Vector Control (TVC)

Description--TVC is required to provide attitude control of the spacecraft during the firing of the midcourse engines. TVC adjusts for thrust misalignments, thrust variations, and variations of the spacecraft c.g. position.

*See D2-82709-10 Classified Supplement - Reference Page 25

The performance of the TVC system must satisfy minimum maneuver and pointing error requirements. Spacecraft rates following engine shutdown must be such that gyro limitations and low level reaction control system capabilities are not exceeded.

Candidate Systems--Three candidate systems were considered:

1) Jet vane control;

2) Gimbaled engine control;

3) Pulsed or throttled engine control.

Combination systems were also examined. They were discarded because of expected complexity, lower reliability, weight and power penalties, and control logic limitations.

Competing Characteristics--Selection of the preferred TVC concept was based on the following competing characteristics.

1) Reliability;

2) Previous space experience;

3) Control dynamics;

4) Availability;

5) Flexibility for growth;

6) Weight.

Selection Rationale and Discussion--Gimbaled engine and pulsed engine control were rejected for the following reasons:

Gimbaled Engine--Monopropellant engines consist basically of a catalyst-filled reactor with propellant inlet valve and an expansion nozzle. The penalties in reliability, weight, power, and flexibility for growth resulting from providing such an engine with a two-degree-of-freedom gimbal structure and associated actuators precluded its selection.

Pulsed-or-Throttled-Engine--These systems appear attractive from weight and control dynamics considerations. However, the reliability of monopropellant engines employing the Shell catalyst in pulsed mode for space application has not yet been characterized fully. The throttled mode introduces complexity and has slower response. Failure modes due to catalyst loss, operation above and below thermal limits, and clogged inlet valves may occur. The pulsed or throttled system is, therefore, rejected on the grounds of low confidence in predicted reliability.

A jet vane TVC system with proven space experience (Mariner II, Mariner IV) offers the design approach with the least technical risk. Good control response of the relatively small vanes can be achieved. This provides sound basis for the design of an autopilot system with satisfactory performance characteristics. The major limitation of the jet vane system is that thrust vectoring capability is limited to about ± 5 degrees. A jet vane TVC system was selected.

4.2.1.4 Engine Multiplicity

Candidate Systems--Single-engine and multiple-engine installations were considered.

Competing Characteristics--Competing characteristics are:

1) Reliability

2) Feasibility

3) Integration with the spacecraft

4) Weight

Selection Rationale and Discussion--Past planetary exploration vehicles (Mariner II and IV) successfully completed their missions with a single-engine installation. A single-engine installation is not feasible on the Voyager spacecraft because:

1) Engine thrust must be applied through the spacecraft c.g.

2) Spacecraft configuration precludes mounting a single monopropellant engine on the spacecraft roll axis.

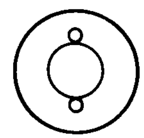
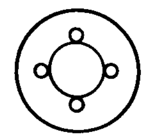
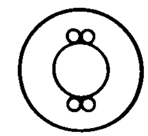
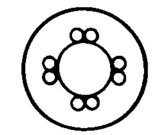
3) The c.g. location has a large excursion along the spacecraft roll axis due to orbit insertion propellant consumption and capsule separation.

4) Jet vane thrust vector control, due to its ±5-degree thrust vector deflection limitation, cannot accommodate the range of longitudinal c.g. shifts when a single engine is mounted off the roll axis.

A multiple-engine installation is selected because the c.g. excursions can be accommodated, using jet vane TVC, by judicious orientation and arrangement of the engines.

4.2.1.5 Number and Location of Engines

Candidate Systems--Two-, four-, and eight-engine configurations were considered. The engines were arranged symmetrically for the two-engine and one of the four-engine configurations. The other four-engine and the eight-engine configurations were arranged in symmetric pairs as shown in Figure 4.2-1.

NUMBER OF ENGINES:	2	4	4	8
Based on 1971 mission, 2000 lb Capsule, 200 lb Thrust Engine, Jet Vane TVC, Engine-Out Capability				
CANT ANGLE (DEGREES)	20.4°	13.5°	13.0°	0°
FAILURE EVENT PROBABILITY (ENGINE):				
Catastrophic Events	0.672 (10^{-6})	0.0034 (10^{-6})	0.0034 (10^{-6})	0.0063 (10^{-6})
Orbit Trim	6.3 (10^{-6})	0.0315 (10^{-6})	0.0315 (10^{-6})	0.0589 (10^{-6})
Midcourse or Orbit Trim	5.37 (10^{-6})	0.0269 (10^{-6})	0.0269 (10^{-6})	0.0501 (10^{-6})
Midcourse and Orbit Trim	2.586 (10^{-6})	0.0129 (10^{-6})	0.0129 (10^{-6})	0.0241 (10^{-6})
SIMPLICITY: (Components, Leak Paths, Manifolding, Signals)	A	C	B	D
PACKAGING	A	C	B	D
POWER REQUIREMENTS	A	B	A	D
THERMAL REQUIREMENTS	A	C	B	D
SERVICE ACCESS	A	C	B	D
PERFORMANCE:				
Minimum Velocity (meters/sec)	0.055	0.11	0.11	0.22
Tolerance (meters/sec)	0.007	0.013	0.013	0.026
Pointing Error (Engine-Out)	23.8°	8.4°	8.0°	3.6°
WEIGHT (POUNDS):				
Hardware	71.7	149.4	131.4	254.7
W_p, Cant + Nominal TVC	135.6	60.9	57.6	5.4
W_p, Gravity Penalty	69.3	4.5	4.5	1.0
W_p, Engine-Out Allowance	48.5	36.3	36.3	24.5
TOTAL WEIGHT	(325.1)	(251.1)	(229.8)	(285.6)

Note:
A Denotes Best System

Figure 4.2-1: Trade Study Summary — Number and Location of Monopropellant Engines

Competing Characteristics--The competing characteristics for selection are:

1) Reliability
2) TVC performance
3) Vehicle-Integration Simplicity
4) Weight

Selection Rationale and Discussion--It is desired that no component failure cause a catastrophic mission failure. Since a multiple-engine installation is required, it is reasonable to provide the selected system with satisfactory single-engine-out capability.

Engine failures that do not result in vehicle destruction may result in unacceptable disturbance torques unless sufficient thrust vector control authority remains to overcome these torques. Control authority is limited by the ±5-degree capability of the jet vanes. Because of:

1) Control authority limitation of the jet vane system,
2) Longitudinal c.g. locations,
3) Configuration constraint on engine location, and
4) The desirability of avoiding malfunction detection equipment,

a canted engine installation is selected. A canted engine installation results in an extended engine life requirement. A propellant weight penalty is also incurred. Minimum cant angle is obtained when the engines are located as far longitudinally from the nominal c.g., but as close laterally as possible to the roll axis.

Cant angle selection is a function of the number of engines and the configuration. Cant angle requirements for static stability (with one engine out) are shown for a typical system in Figure 4.2-2. The engine cant angle requirement is selected for the spacecraft condition resulting in the largest d/ℓ ratio (perpendicular distance of jet vane from the pitch-plane divided by distance between jet vane station and c.g. location). The maximum ratio occurs at orbit trim with capsule off. An additional selection constraint is the line of maximum allowable jet vane deflection, δ_T. For the systems considered, the 5-degree maximum is reduced by a 1-degree contingency for dynamic stability, and the effects of tolerances in the spacecraft lateral c.g. offsets, thrust variations, and engine misalignments. This constraint is denoted by the inner dashed line for the +4-degree thrust vector angle. The minimum d/ℓ capability of the system is shown by the inner dashed line for the -4-degree thrust vector angle. The range of spacecraft d/ℓ ratios must be between these two dashed lines. The minimum cant angle in this range can then be selected. Should the range of d/ℓ ratios fall outside one of the dashed lines, an alternate engine configuration must be selected. The range of d/ℓ ratios for the 1971 mission spacecraft with 2000-lb capsule can be accommodated only marginally by a two-engine system. Figure 4.2-3 shows the constraining d/ℓ lines and the corresponding δ_T lines for the candidate systems. Cant angle and other competing characteristics are summarized in Figure 4.2-1.

The two-engine system is rejected on the basis of:

1) Marginal static stability with an engine out;
2) High probability of catastrophic mission failure.
3) Large engine-out pointing error (due to large cant angle);
4) High weight (due, in part, to large cant angle).
5) Lack of growth capability.

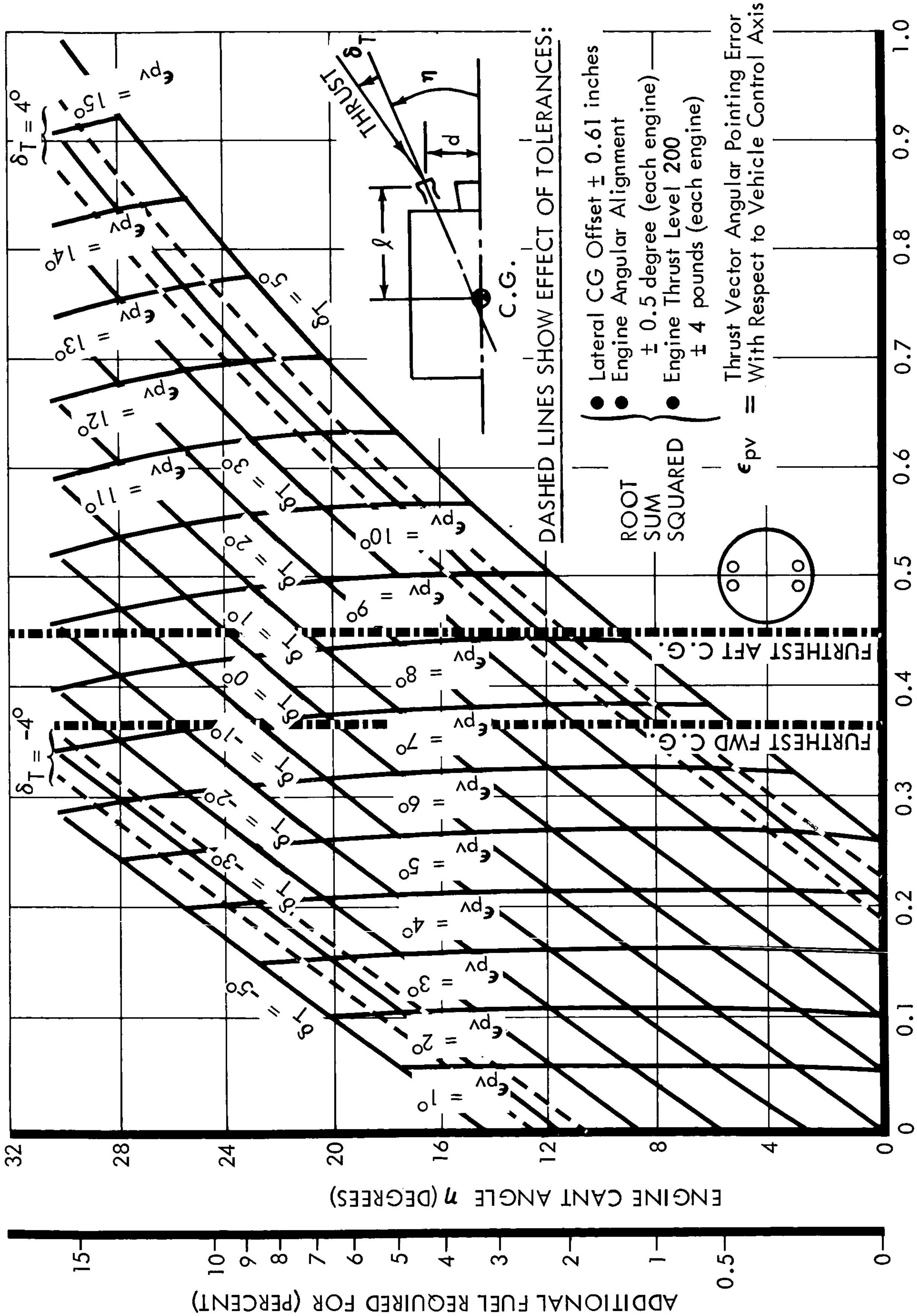


Figure 4.2-2: Static Stability Requirements With One Engine Failure — Four-Engine Configuration with Engines Paired

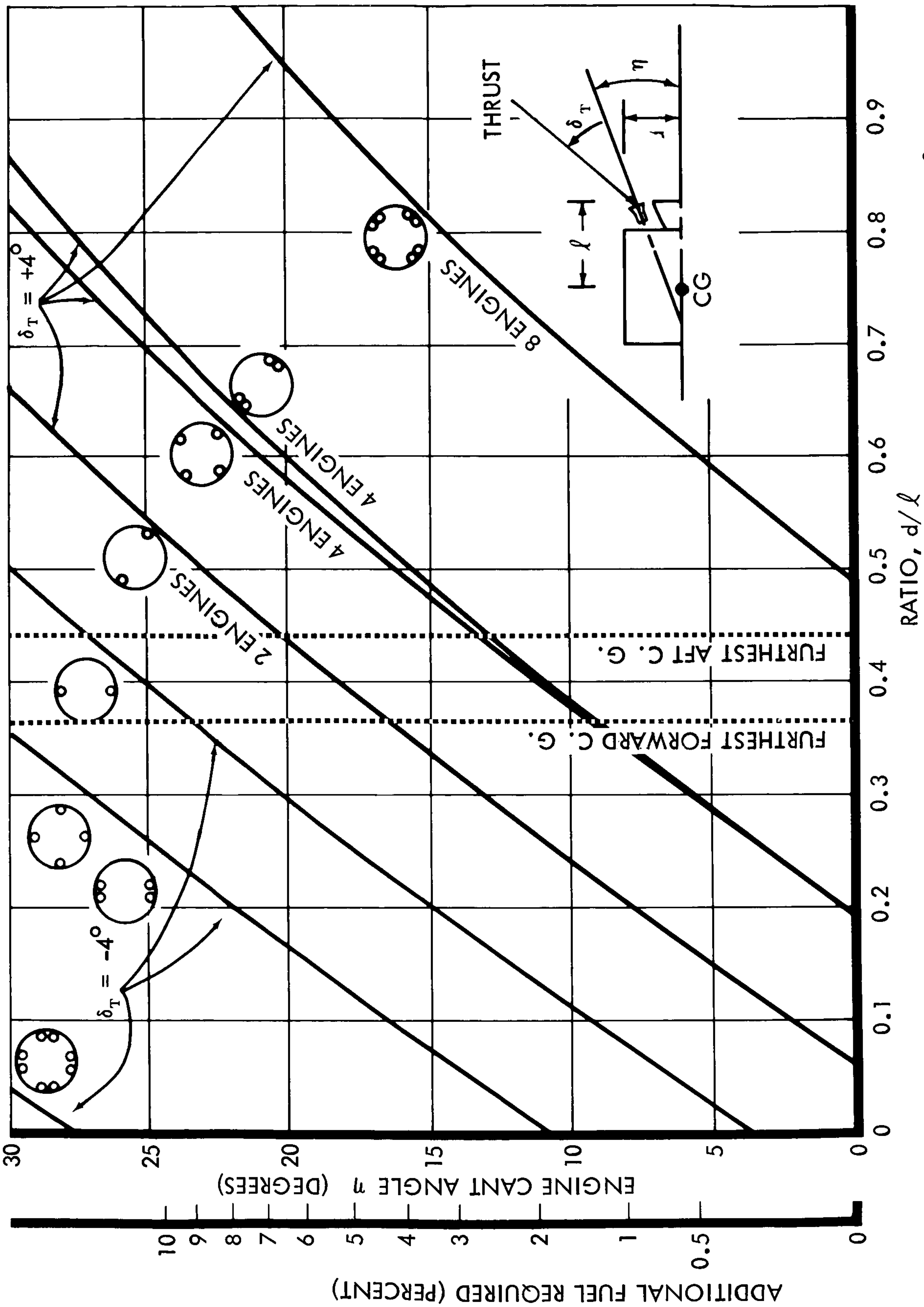


Figure 4.2-3: Candidate Systems Static Stability Capability for $\delta_T = \pm 4^\circ$ One Engine Out

The four quadrant symmetrical four-engine system was rejected on the basis of:

1) High weight;

2) Vehicle integration problems (thermal, servicing, packaging).

The eight-engine system was rejected on the basis of:

1) Complexity;

2) High catastrophic-engine-failure probability (more leak paths);

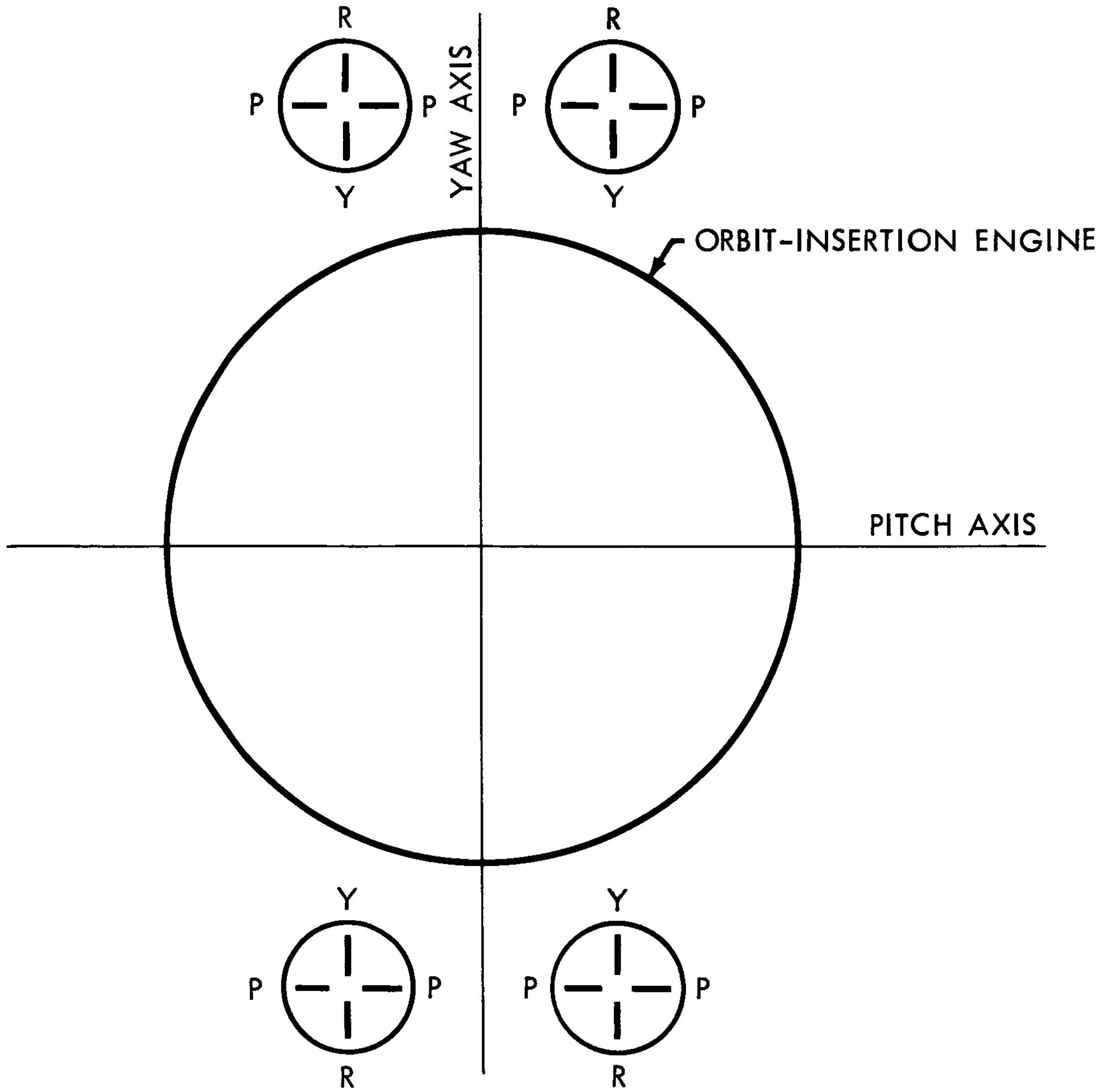
3) High weight (hardware);

4) Vehicle integration problems (thermal, servicing, packaging).

The selected system with two pairs of engines placed symmetrically about the pitch and yaw axes shows reliability and weight advantages over the alternate systems. An additional advantage is obtained in yaw control over the other four-engine system. The d/ℓ ratios for yaw are much reduced over those for pitch. Static stability (one engine out) can be provided by a 2-degree thrust vector deflection with uncanted engines. However, the engines will be canted so that all thrust lines lie along the sides of a right circular cone with the apex on the roll axis providing a yaw cant angle of 2.6 degrees. The required control authority with one engine out is obtained by using half the number of jet vanes for yaw control as that required for pitch control (Figure 4.2-4). This mechanization results in a control system that does not require mixing of pitch and yaw signals with roll signals. Autopilot complexity is reduced (especially in redundancy).

4.2.1.6 Engine Thrust

Candidate Systems--Engines in the 50- to 800-lb thrust range were considered.



P PITCH CONTROL (8 VANES)
Y YAW CONTROL (4 VANES)
R ROLL CONTROL (4 VANES)

PITCH CONTROL AUTHORITY EQUALS
TWICE YAW CONTROL AUTHORITY

Figure 4.2-4: Jet-Vane Thrust-Vector-Control Logic

Competing Characteristics--The following characteristics were considered:

1) Engine availability;

2) Engine burn time;

3) Propellant penalties during orbit trim and with engine out;

4) Engine system weight;

5) Growth and flexibility requirements.

Selection Rationale--Velocity-increment penalties for orbit trimming at periapsis in a typical Mars-bound orbit are shown in Figure 4.2-5 as a function of initial thrust-to-weight ratio. Additional propellant penalties are associated with engine-out operation under these conditions. Figures 4.2-6 and 4.2-7 show the propulsion system weights which are sensitive to thrust for the preferred four-engine configurations for 1971 and 1975 missions, with and without engine-out effects. Minimum thrust levels established by maximum-maneuver-time limitations are also indicated on these figures. The thrust-sensitive weight increment contained in the curves consists of: engine, valve, thrust vector control system, thrust mount, thermal control, and propellant weight. Minimum-weight systems for 1971 missions require thrust levels lower than those allowed by maneuver-time limits. Those for 1975 missions require thrust levels slightly in excess of those allowed by maneuver time limits.

No existing monopropellant engines are available in the desired thrust range. Also, a new engine is necessary to take advantage of the selected spontaneous catalyst. A new engine of 200-pound thrust is therefore selected

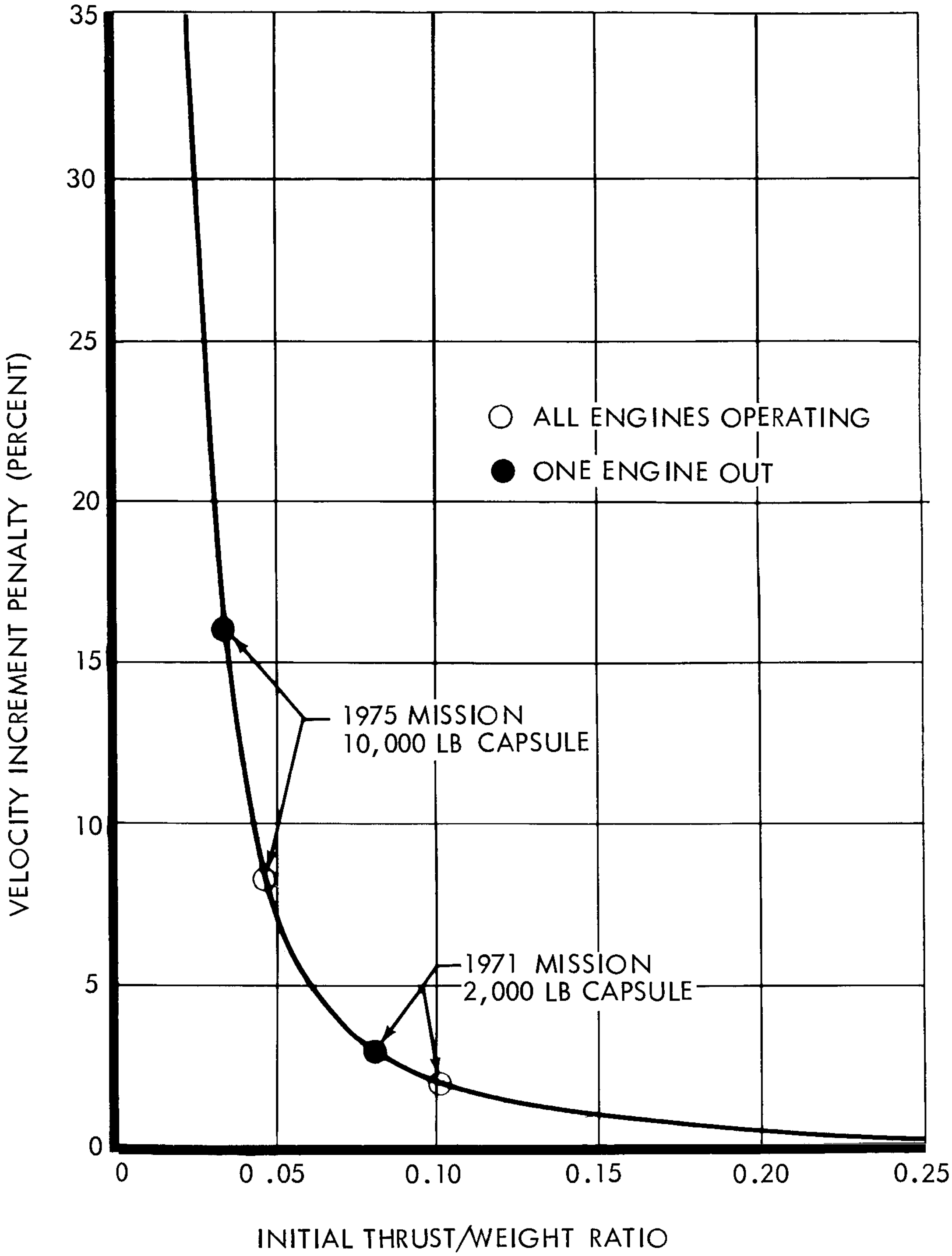


Figure 4.2-5: Orbit Trim Velocity Penalty

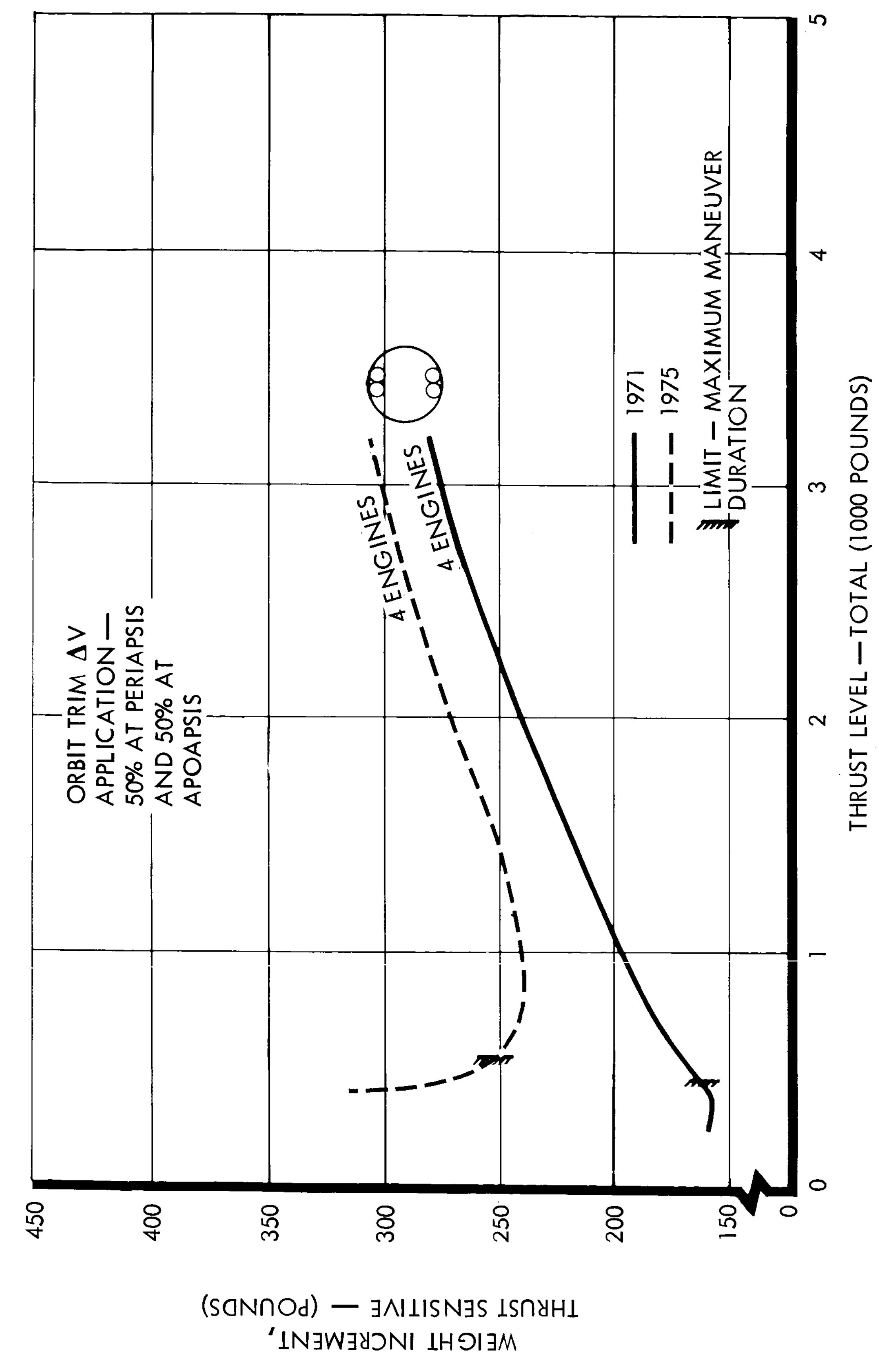


Figure 4.2-6: Monopropellant Engine Thrust Level — Weight Trade Study Without Engine-Out Allowances

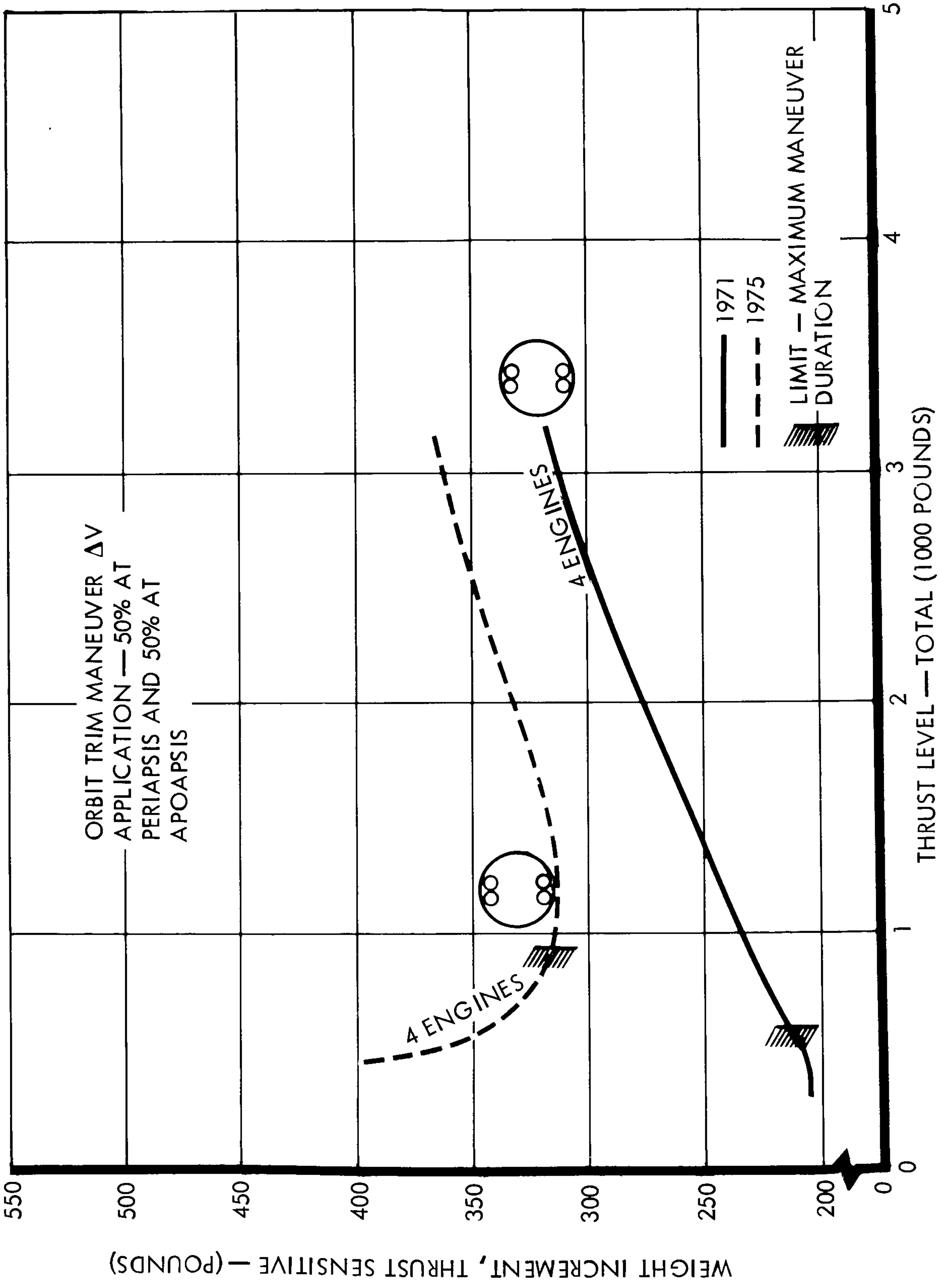


Figure 4.2-7: Monopropellant Engine Thrust Level — Weight Trade Study With Engine-Out Allowances

for this application on the basis of (1) application to all Voyager missions through 1977, (2) overall weight, and (3) maneuver-time limitations. A 200-pound thrust hydrazine engine has been fabricated successfully by JPL.

4.2.1.7 Monopropellant Engine Summary

The selected system is a symmetrically paired four-engine configuration using 200-lb engines. The engines are canted 13 degrees in the pitch plane, 2.6 degrees in the yaw plane, and have $\pm$ 5-degree thrust vector capability provided by jet vane deflection. Uncoupled control is achieved using the vane arrangement shown in Figure 4.2-4. A representative autopilot block diagram is shown in Figure 4.2-8. Typical start burn time responses, assuming perfect sensors, are shown in Figure 4.2-9.

4.2.1.8 Expulsion

Candidate Methods--Positive expulsion devices considered for the monopropellant system were:

1) Butyl rubber bladders;
2) Teflon bladders;
3) Convoluted metal diaphragms;
4) Rolling metal diaphragms;
5) Metal bellows.

Competing Characteristics--The following competing characteristics were considered in the final selection of the preferred expulsion method:

1) Reliability;
2) Compatibility with hydrazine;
3) State of development;
4) Previous space usage

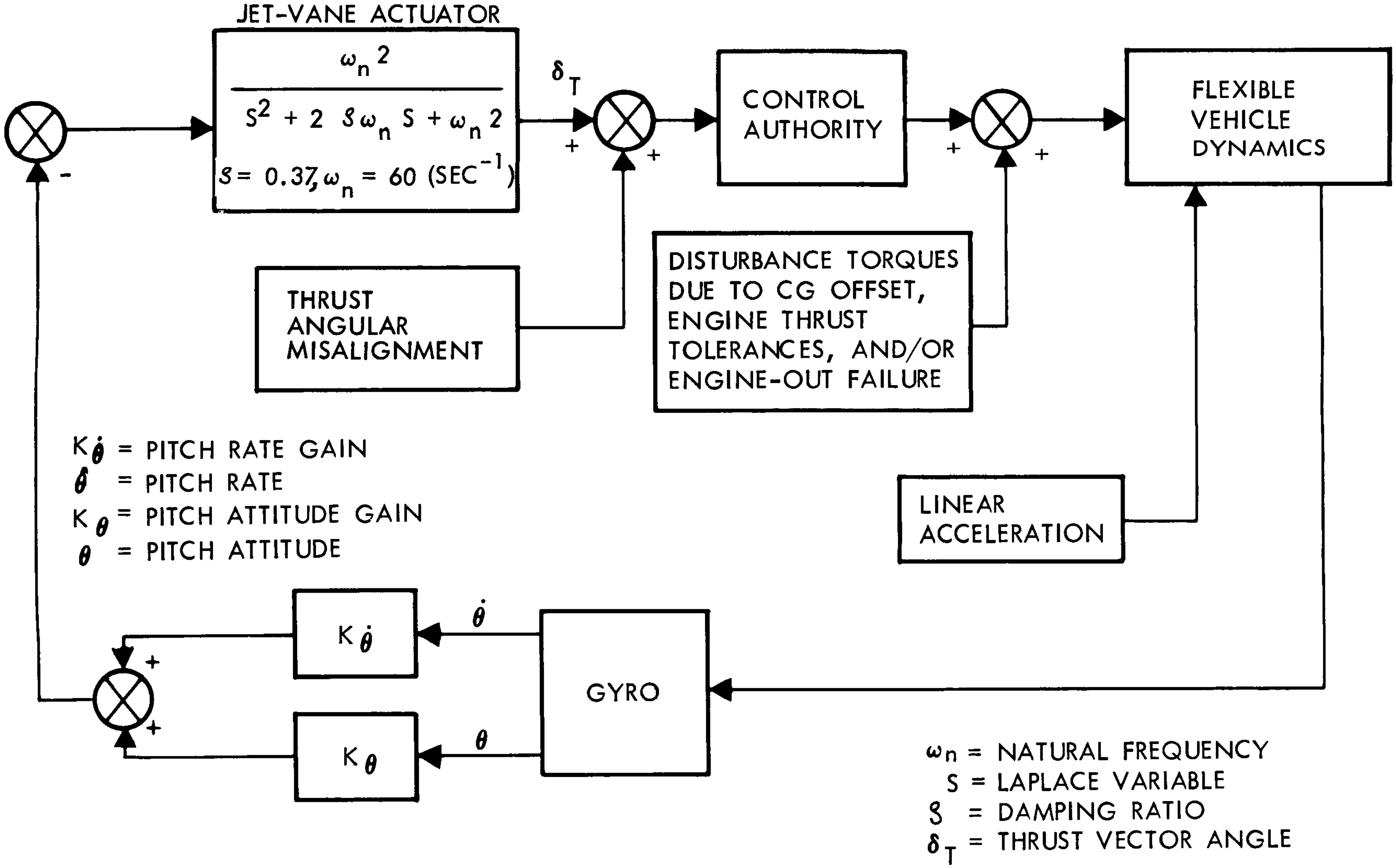


Figure 4.2-8: Jet-Vane Thrust-Vector-Control Autopilot (Pitch, Yaw, & Roll Axes)

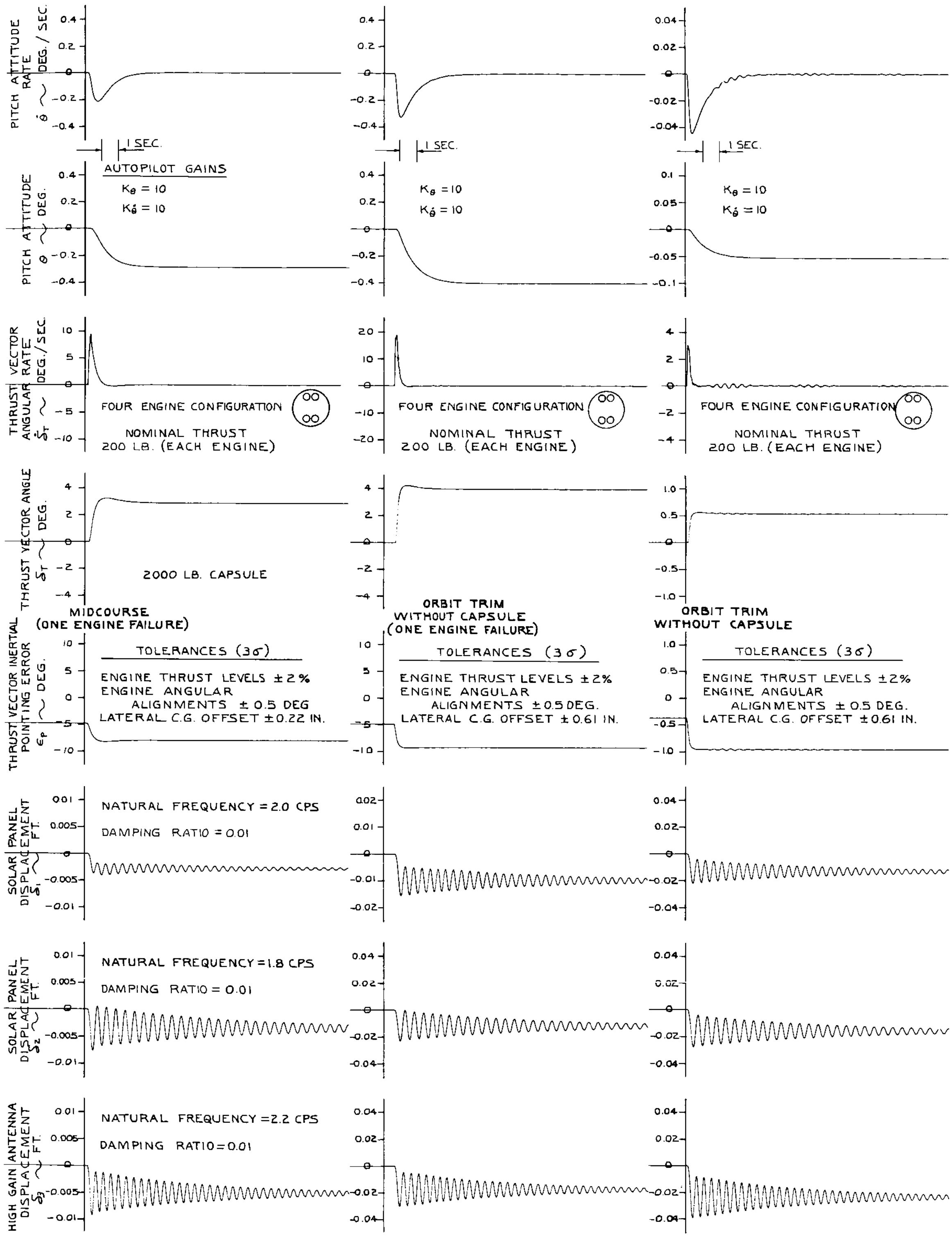


Figure 4.2-9: Monopropellant Subsystem Start Transients — Jet-Vane TVC

5) Permeability;

6) Cycling capability;

7) Weight.

Selection Rationale--Physical and operating characteristics of the expulsion methods are presented in Table 4.2-1. Butyl rubber bladders were selected for the preferred design because their reliability has been high on previous spacecraft, including Ranger and Mariner. Butyl rubber bladders are lightweight, nonpermeable, can be cycled many times, and are only slightly reactive with hydrazine. The expulsion efficiency of this device is high.

The teflon bladder was rejected because of a high rate of permeation. It allows pressurant gas to enter the fuel side of the tank, and results in increased pressurant gas consumption and erratic engine operation.

Both the convoluted and rolling metal diaphragms show promise for future use. Neither is considered sufficiently developed to provide the high reliability required for the Voyager mission. In addition, the single cycle of operation penalizes tank and system inspection and checkout.

The metal bellows is a reliable device. It is compatible with the fuel, and is capable of many cycles. It was rejected because it is the heaviest method consider, has poor volumetric efficiency, and does not offer a significant reliability gain.

Table 4.2-1: MONOPROPELLANT SYSTEM EXPULSION METHOD SUMMARY

	Expulsion Methods				
	Butyl Bladders (Collapsing)	Teflon Bladders (Collapsing)	Metal Convoluted Diaphragms	Metal Rolling Diaphragms	Metal Bellows
Expulsion efficiency, percent	98	98	97	98	96
Reliability	Good	Poor	Poor	Poor	Good
Cycle life (complete expulsions)	200	50	1	1	200
Permissible tank shape	Sphere/Cyl.	Sphere/Cyl.	Spher.	Cyl.	Cyl.
Maximum dimension, inches	60	60	60	20	20
Specific weight ratio	Good	Good	Fair	Good	Poor
Volumetric efficiency	Good	Good	Good	Fair	Poor
Pressure drop, psi	3	3	20-100	15-35	10
Expulsion pressure (high or low)	Both	Both	Both	Both	Both
Inherent c.g. control (multiple tanks)	Good	Good	Poor	Poor	Fair
Simplicity	Good	Good	Fair	Fair	Good
Shelf life	Fair	Fair	Good	Good	Good
Mission life	Good	Short	Good	Good	Good
Development cost	Low	Mod.	High	High	Low
Fabrication cost	Low	Mod.	High	High	High
Testing cost	Low	Low	High	High	Low
Permeability	Low	High	None	None	None
Capability for sterilization	Fair	Fair	Yes	Yes	Yes
Compatibility with N_2H_4	Fair	Good	High	High	High
Development time	Short	Short	Long	Long	Short
Ease of fabrication	Fair	Poor	Poor	Poor	Poor
Magnetic compatibility	Good	Good	Good	Good	Fair
Acceptance testing feasibility	Yes	Yes	No	No	Yes
Ease of propellant loading	Fair	Fair	Poor	Fair	Fair
Checkout feasibility	Yes	Yes	No	No	Yes
Replaceability	Good	Good	Poor	Poor	Good
Radiation sensitivity	Fair	Good	Good	Good	Good

4.2.1.9 Propellant Storage

Selection of butyl bladders as the expulsion device in the propellant tanks permits use of either spherical or cylindrical tankage. Tankage cannot be located on the vehicle longitudinal centerline because of the solid orbit-insertion motor. An even number of tanks must therefore be installed to minimize c.g. travel. Available space in the vehicle requires the installation of four cylindrical tanks with hemispherical ends to contain the required 2495 pounds of usable hydrazine. A slight weight penalty is incurred by selecting cylindrical over spherical tanks.

Center-of-gravity control of the spacecraft propulsion system during propellant expulsion is maintained by the inherent ability of the bladders to expel fuel at a low pressure differential. Movement of fuel between tanks due to sloshing or TVC dynamics is damped by interconnecting plumbing.

4.2.1.10 Pressurization

Candidate Pressurant Systems--The pressurization systems considered for the monopropellant propulsion system were:

1) Stored nitrogen;
2) Stored helium;
3) Gas Generator;
4) Blowdown system;
5) Use of reaction-control-system nitrogen storage tankage.

Competing Characteristics --The following characteristics were considered in the final selection of the preferred pressurization system.

1) Reliability;
2) State of development;
3) Leakage;
4) Modular packaging;
5) Weight.

Selection Rationale--Table 4.2-2 is a summary of the physical and operating characteristics of the candidate pressurizing systems. The stored nitrogen system was selected as the preferred system because of previous space experience (Mariner and Ranger) and low leakage rates. The self-contained concept allows for modular packaging.

The stored-helium system was the lightest gaseous system considered. It was rejected because of its greater leakage rates, and limited previous usage.

Tank pressurization by means of a hydrazine gas generator has attractive characteristics. Pressurant is stored in liquid form and is converted into gas with the aid of the new Shell 405 catalyst. A system with fuel being obtained from the monopropellant system tankage is also feasible. This concept was not selected because of lack of space experience.

Blowdown-system weight was excessive because of the increase in wall thickness required by the increased tank pressure. This system requires installation of an undeveloped liquid regulator, if thrust level control is desired.

Storage of the pressurizing gas in the reaction-control (RC) system tankage was rejected because it: (1) compromises the reliability of both the RC and liquid propulsion systems and (2) prevents the development of a modular concept for both systems.

Table 4.2-2 MONOPROPELLANT PRESSURIZATION SYSTEM TRADES

PERFORMANCE PARAMETERS*	PRESSURIZATION CONCEPT				
	Nitrogen (N_2)	Helium (He)	Blowdown System (N_2)	Gas Generator (N_2H_4)	RC System Tankage (N_2)
Fuel Tank Pressure, psi	280	280	3500 to 300	280	280
Gas Tank Pressure, psi	3500	3500	3500 to 300	280	3500
Pressurant Gas Wt, Lb	65	9.3	65	(Liquid N_2H_4 31.2)	65
Pressurant Tank We, Lb (Spherical)	142	146	(Fuel and Presst. 720)	(System) 67.5	142
Total Wt, Gas + Gas Tank, + Propellant Tank-Lb	417	365.3	785	308.7	417
Relative Leakage Rate	1	2.65	1	1	1
Solubility in N_2H_4,% by Wt	0.065	0.0062	0.065		0.065

* Fuel Quantity = 2495 lb
2000-lb Capsule

4.2.1.11 Isolation Valving and Plumbing

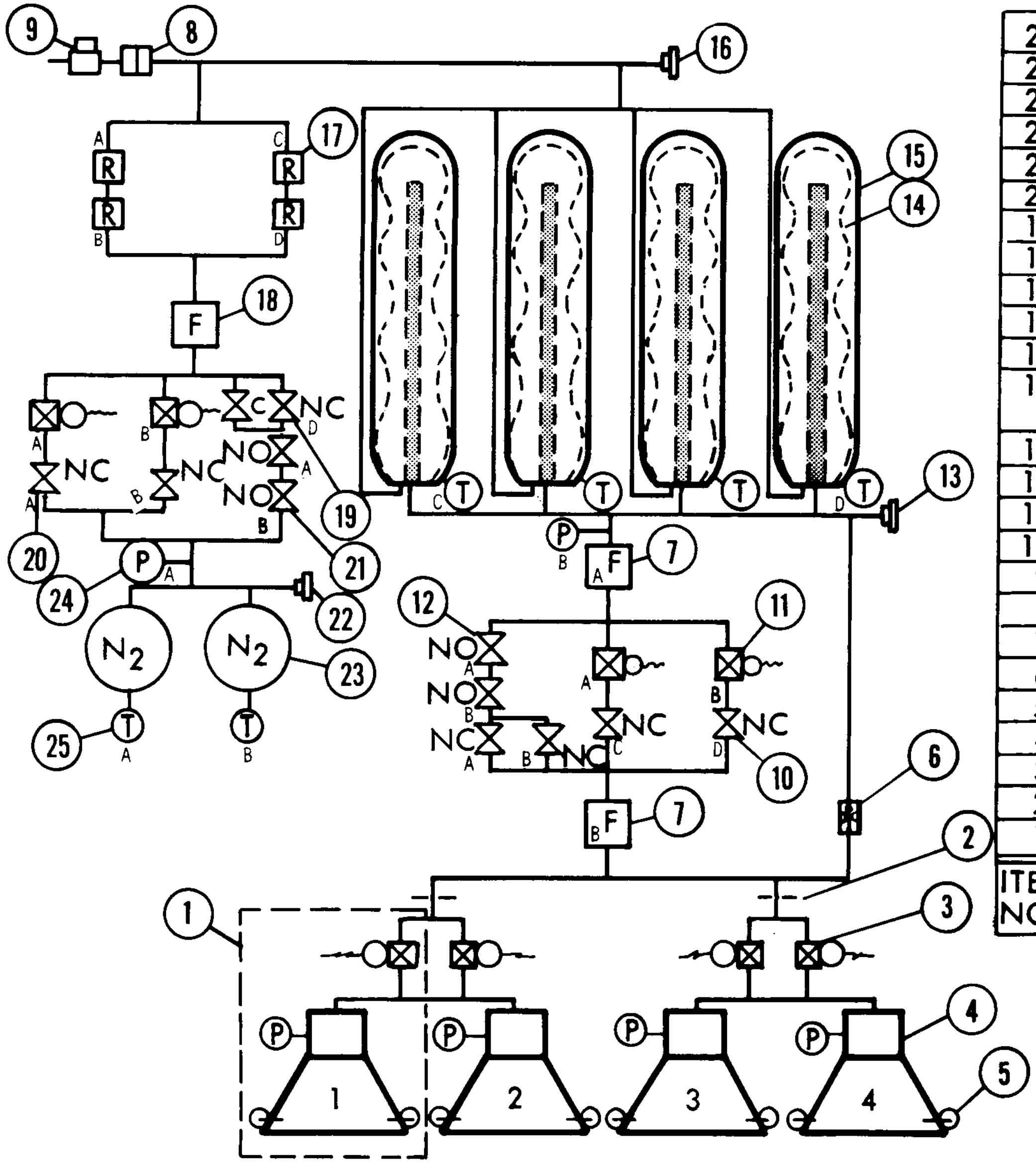
Candidate Fluid Systems--The fluid systems considered for use in the monopropellant propulsion system are presented in Figures 4.2-10, 4.2-11, and 4.2-12. All three systems include four cylindrical storage tanks using butyl rubber bladders for expulsion. A self-contained nitrogen-tank pressurization system is included, with two spherical tanks for storage. The three systems differ in the amount of redundancy provided in flow-control devices.

Competing Characteristics--The following characteristics were considered in the final selection of the preferred fluid system:

1) Reliability;
2) Zero leakage during long-term shutdown;
3) Absence of catastrophic failure modes;
4) Simplicity;
5) Weight.

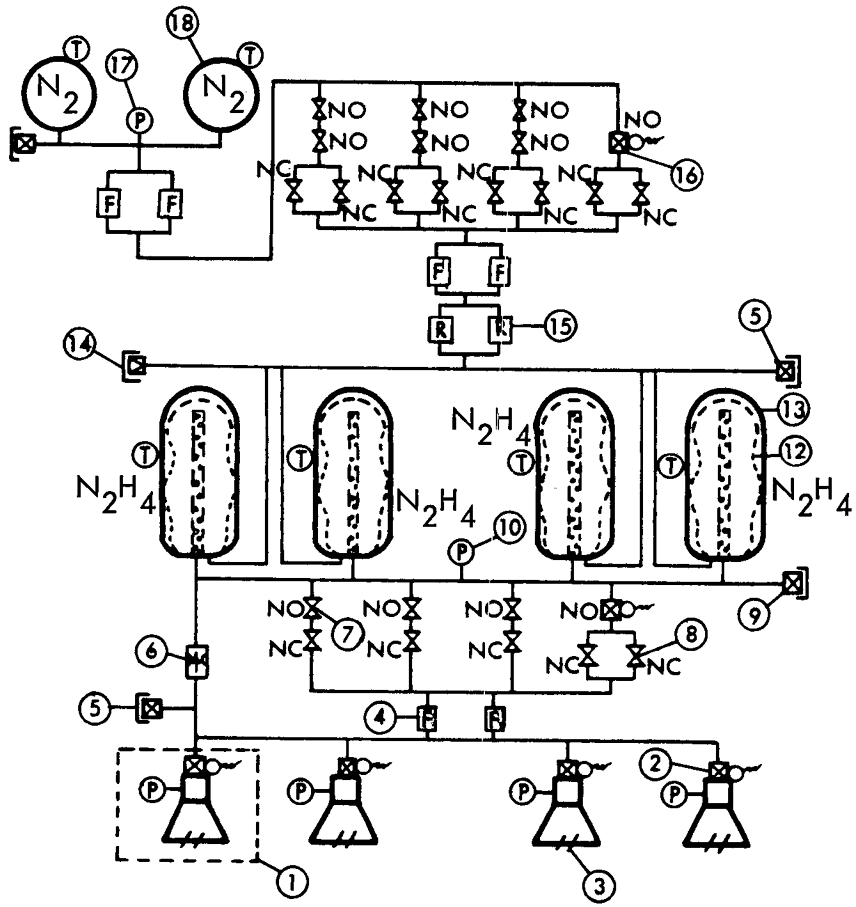
Selection Rationale--Table 4.2-3 is a summary of the fluid-system characteristics that were considered. System A, shown in Figure 4.2-10, was selected as the preferred system for the following reasons.

1) Sufficient operating paths are provided to perform the required propulsive maneuvers. Redundant shutoff valves are installed in such a manner that no single control device failure can cause catastrophic failure of the mission. This is considered important.
2) Overall system reliability is high.
3) The system can be maintained by closed squib valves in a zero-leakage condition until the second midcourse correction maneuver.



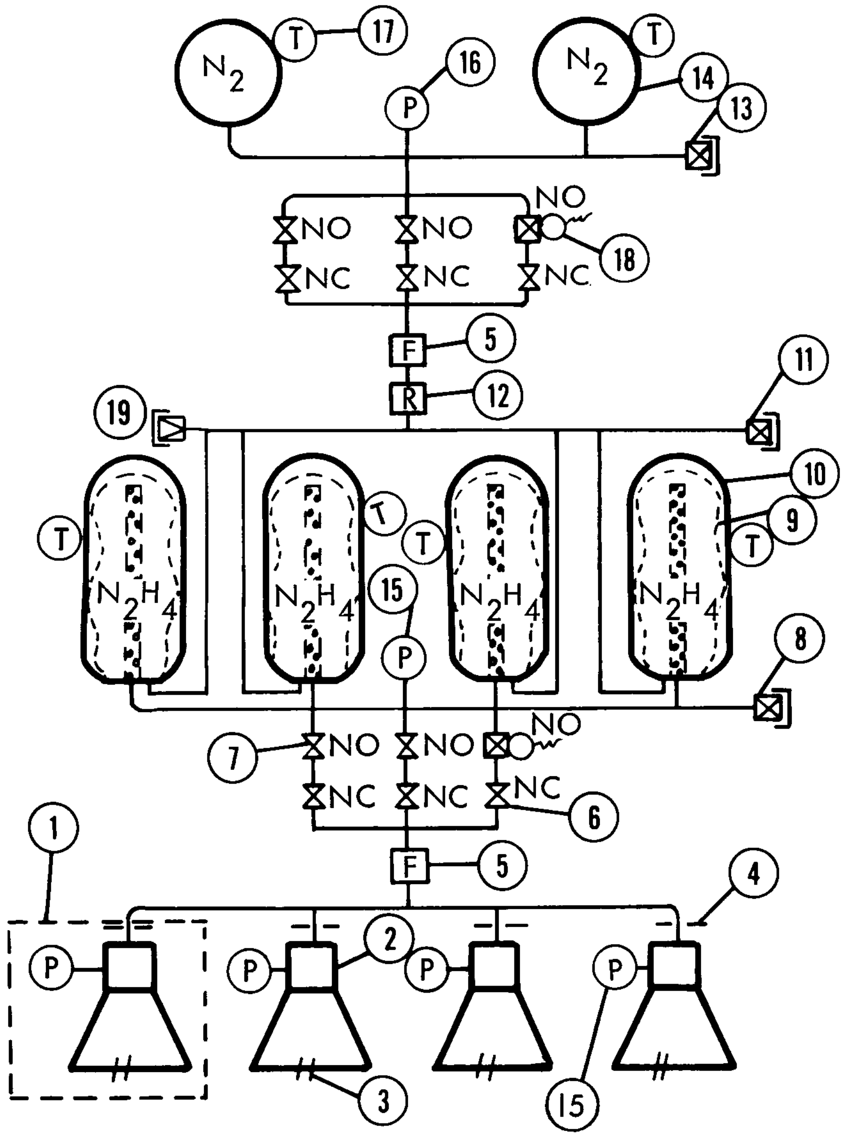
ITEM NO.	QTY.	ITEM
25	4	TRANSDUCER, TEMPERATURE
24	6	TRANSDUCER, PRESSURE
23	2	TANK, NITROGEN
22	1	VALVE & CAP, PRESSURE
21	4	VALVE, SQUIB, N.C.
20	2	VALVE, LATCHING SOLENOID
19	2	VALVE, SQUIB, N.O.
18	1	FILTER, NITROGEN
17	4	REGULATOR, NITROGEN
16	1	VALVE & CAP, VENT & PRESSURE
15	4	TANK, PROPELLANT
14	4	COLLAPSING BLADDER & STANDPIPE POSITIVE EXPULSION
13	1	VALVE & CAP, FILL & DRAIN
12	2	VALVE, SQUIB, N.O.
11	2	VALVE, LATCHING SOLENOID
10	4	VALVE, SQUIB, N.C.
9	1	VALVE, RELIEF
8	1	BURST DISC
7	2	FILTER, PROPELLANT
6	1	THERMAL RELIEF
5	16	JET VANE & ACTUATOR ASSY
4	4	CATALYST BED
3	4	VALVE, LATCHING SOLENOID
2	2	ORIFICE
1	4	ROCKET ENGINE ASSY

Figure 4.2-10: Monopropellant Midcourse And Orbit Trim — Isolation Valving And Plumbing (System A)



ITEM NO.	QTY.	ITEM
18	2	TANKS, NITROGEN STORAGE
17	1	TRANSDUCER, PRESSURE (HIGH)
16	2	VALVE, SOLENOID, NORMALLY OPEN
15	2	VALVE, PRESSURE REGULATOR
14	1	VALVE, PRESSURE RELIEF
13	4	TANK, FUEL STORAGE
12	4	COLLAPSING BLADDER & STANDPIPE POSITIVE EXPULSION
11	6	TRANSDUCER, TEMPERATURE
10	5	TRANSDUCER, PRESSURE (LOW)
9	1	VALVE, FILL & DRAIN, LIQUID
8	13	VALVE, SQUIB, NORMALLY CLOSED
7	9	VALVE, SQUIB, NORMALLY OPEN
6	1	VALVE, THERMAL RELIEF
5	2	VALVE, VENT & TEST
4	6	FILTER
3	16	VANE, JET ASSY
2	4	VALVE, PROPELLANT SOLENOID
1	4	ENGINE, ROCKET ASSY

Figure 4.2-11: Monopropellant Midcourse And Orbit Trim — Isolation Valving And Plumbing (System B)



ITEM NO.	QTY	ITEM
19	1	VALVE, PRESSURE RELIEF
18	2	VALVE, SOLENOID, LATCHING
17	6	TRANSDUCER, TEMPERATURE
16	1	TRANSDUCER, PRESSURE (HIGH)
15	5	TRANSDUCER, PRESSURE (LOW)
14	2	TANK STORAGE, NITROGEN
13	1	VALVE, FILL, GAS
12	1	VALVE, REGULATOR
11	1	VALVE, VENT
10	4	TANK, STORAGE, FUEL
9	4	COLLAPSING BLADDER & STANDPIPE POSITIVE EXPULSION
8	1	VALVE, FILL, LIQUID
7	4	VALVE, SQUIB, NORMALLY OPEN
6	6	VALVE, SQUIB, NORMALLY CLOSED
5	2	FILTER
4	4	ORIFICE, TRIM
3	4	VANE, JET ASSY
2	4	CATALYST BED
1	4	ENGINE, ROCKET ASSY

Figure 4.2-12: Monopropellant Midcourse And Orbit Trim — Isolation Valving And Plumbing (System C)

Table 4.2-3: ISOLATION VALVING AND PLUMBING SYSTEMS COMPARISON--MONOPROPELLANT SYSTEM

CHARACTERISTICS	SYSTEM		
	A	B	C
Number of Components	58	51	28
System Weight, lb	371.0	367.7	344.5
Total Number of Connections	180	226	110
System Reliability	0.998197	0.998228	0.998029
Redundant Components			
Pressure regulators	3	1	0
Filters	0	3	0
Relief valves	0	0	0
Squib valves	8	8	4
Solenoid valves	8	4	0
Number of Shutoff Valves			
Squib	12	24	10
Solenoid	12	6	2
Manual	3	3	3
Loss of System Failure Modes	0	2	6

4) Filling of the fuel and pressurant tanks, and system checkout can be accomplished simply.

5) System weight is acceptable.

The preferred system (Figure 4.2-10) uses brazed and welded connections between all fittings, components, and tubing. With careful fabrication and inspection techniques, an essentially zero-leakage system is provided. The filters installed in both the liquid and gaseous portions of the system are of sufficient capacity to provide high reliability. Nitrogen-pressure regulators are of the type used on the Mariner program, providing reliable flight-proven hardware.

System B (Figure 4.2-11) was found to be the most reliable of those considered. It provides positive isolation until the fourth liquid propulsion maneuver. This was not considered essential because the time between the second and fourth liquid system propulsion maneuvers is relatively short, and its increased number of components and weight is not justified by the slight increase in reliability.

System C (Figure 4.2-12) was attractive because of its simplicity, light weight, and minimum number of components. It was rejected, however, because a single failure of a flow-control device could cause catastrophic failure of the vehicle.

4.2.2 Bipropellant System

4.2.2.1 Propellant, Engine, and Thrust Level

Discussion--Applicable bipropellant engines were discussed in detail in Task A. Candidate engines were limited to those already funded to minimize cost and development time. An adequate number of candidate engines were available in the thrust range of interest. Bipropellant engines considered in this study are described in Table 4.2-4 below.

TABLE 4.2-4

ENGINE	MFG.	CONTRACTING AGENCY	APPLICATION	THRUST (VAC) (LBS)	STATUS
MA-109	Marquardt	NASA	Apollo Lunar Orbiter	100	In Qual
C-1	Thiokol- RMD	NASA	Common Engine	100	In Dev.
8374	Bell-Aero-systems	NASA	Experimental Auxiliary Engine	100	In Dev.
MIRA-180	STL	NASA	Surveyor Back-up	180	Canc.
Beryllium	Rocketdyne	In-house	Not Designated	100, 200	Company Dev.

Competing Characteristics--The following competing characteristics were considered in the final selection of the preferred bipropellant engine:

1) Status and availability;
2) Space-use experience;
3) Thrust level;
4) Engine lifetime.

Selection Rationale--Each of the engines was evaluated against total vehicle thrust and total impulse requirements. The STL MIRA-180 system requires no less than seven engines for 1971 missions and nine for 1975 missions because of its limited operating life. The remaining engine installations were similar in weight except for the proposed Rocketdyne 200-pound thrust engine. For the latter engine, the higher engine thrust level resulted in fewer engines and somewhat less weight. Experience and status of the Marquardt MA-109 engine is a definite advantage in its favor. Consequently, this engine was selected as the preferred bipropellant engine.

As shown in Figure 4.2-13, no fewer than four MA-109 engines must be used in the 1971 and 1973 missions, and six in the 1975 and 1977 missions, due to maximum-maneuver-time limitation.

A four-engine cluster of the MA-109 engine was selected since it represents a system of minimum weight and complexity. The 1975 and 1977 missions will create a requirement for either an eight-engine installation (a six-engine one has undesirable control characteristics) or a relaxation of the thrusting duration limits prescribed for maximum maneuver times.

4.2.2.2 Bipropellant Engines--Installation

Based on considerations identical to those given the monopropellant engines (see Section 4.2.1.5), the four MA-109 bipropellant engines are arranged in opposing pairs of engines.

4.2.2.3 Bipropellant Engines--Thrust Vector Control

Description--Jet vane control is not feasible with bipropellant systems because of higher engine exhaust temperature characteristics. Thrust vector control methods considered were therefore limited to gimbaled engines, differential throttling, and differential engine pulsing.

Competing Characteristics--Major competing characteristics considered were:

1) Reliability;

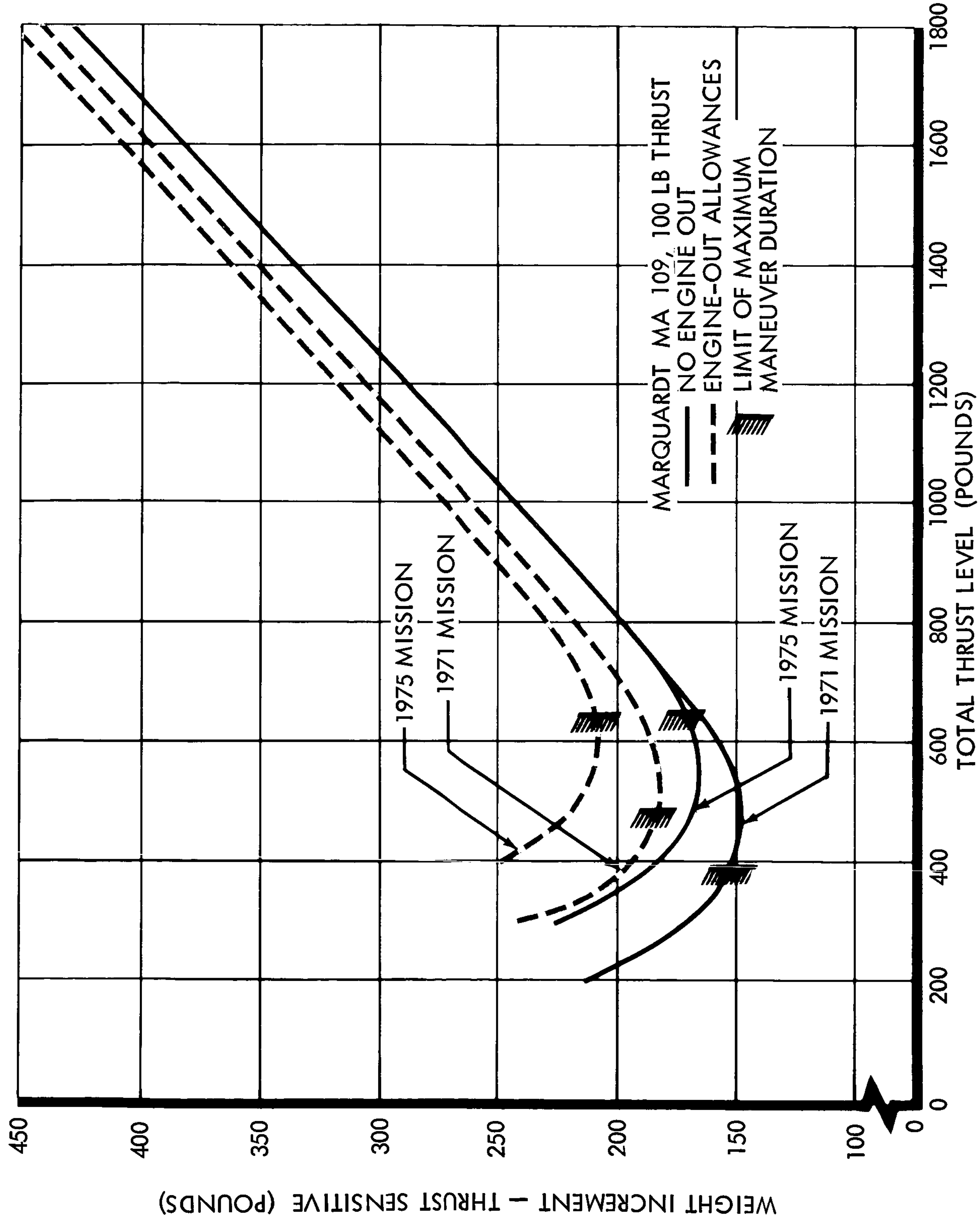


Figure 4.2-13: Engine-Weight Trade Study — Bipropellant Engines

2) Experience;
3) Compatibility with spacecraft dynamics;
4) Weight.

Selection Rationale--As in Task A, the pulsed-midcourse-engine concept was rejected on the basis of the reliability degradation associated with pulsing the MA-109 engines. The gimbaled engine concept with the engines canted at 13 degrees and provided with ±5-degrees gimbal angle capability was selected on the basis of current experience, minimum complexity, adequate control authority, and engine-out capability.

4.2.2.4 Expulsion and Propellant Storage

Candidates--Expulsion methods considered for propellant storage tanks in the bipropellant midcourse propulsion system were limited to all metal devices. This prevents catastrophic failures that may occur when bipropellant fuel and oxidizer are brought together either through permeation or leakage. The metal expulsion devices considered were metal bellows and convoluted metal diaphragms.

Competing Characteristics--The following competing characteristics were considered in the final selection of the tank expulsion device:

1) Reliability;
2) State of development;
3) Cycling ability;
4) Expulsion efficiency;
5) Volumetric efficiency;
6) Expulsion pressure.

Selection Rationale--Physical and operating characteristics of the two expulsion methods are shown on Table 4.2-5. The metal bellows is the heaviest of the two methods considered and its volumetric efficiency in the cylindrical tanks is poor. Metal bellows were selected, however, as the preferred expulsion method because:

1) The reliability of the bellows meets the requirements of the bipropellant propulsion system.
2) Multiple-cycling capability of the bellows provides means of tank and system checkout and inspection.
3) Expulsion pressure of the bellows is relatively low, providing inherent c.g. control during propellant usage.

The convoluted metal diaphragms can be installed in lightweight spherical tanks. The volumetric efficiency of this device is high. The metal diaphragm was rejected, however, because of lower reliability, lack of recycle capability, and lack of flight experience.

Propellant Storage--Trade studies were not conducted on the bipropellant storage tankage. The selection of the metal-bellows expulsion device resulted in the choice of cylindrical tanks. Space available for propellant storage on the vehicle required that four tanks be used.

4.2.2.5 Pressurization

Candidates--Pressurization methods considered for the bipropellant tanks were:

1) Stored nitrogen;
2) Stored helium;

Table 4.2-5: BIPROPELLANT EXPULSION TRADES

	Expulsion Device	
	Metal Bellows	Convoluted Metal Diaphragms
Expulsion efficiency, percent	96	97
Reliability	High	Low
Cycle life (complete expulsions)	200	1
Permissible tank shape	Cylindrical	Spherical
Maximum size, in.	20-in, diam	60-in. diam
Specific weight ratio	Poor	Fair
Volumetric efficiency	Poor	Good
Pressure drop, psi	10	20-100
Expulsion pressure (high or low)	Both	Both
Inherent c.g. control (multiple tanks)	Fair	Poor
Simplicity	Good	Fair
Shelf life	Good	Good
Mission life	Good	Good
Development cost	Low	High
Fabrication cost	High	High
Testing cost	Low	High
Permeability	None	None
Compatibility with sterilization	High	High
Compatibility with Aerozine 50	Good	Good
Compatibility with N_2O_4	Good	Fair
Development time	Short	Long
Ease of fabrication	Poor	Poor
Magnetic compatibility	Fair	Good
Acceptance testing	Yes	No
Ease of propellant loading	Good	Poor
Checkout	Yes	No
Replaceability	Good	Poor
Radiation sensitivity	Good	Good

3) Gas generator;

4) Blowdown system;

5) Use of reaction-control system nitrogen bottles.

Competing Characteristics--The following competing characteristics were considered in the final selection of the preferred pressurization system:

1) Leakage characteristics;

2) Weight;

3) Status of development;

4) Reliability;

5) Ease of packaging.

Selection Rationale--The self-contained nitrogen system was selected as the preferred system because:

1) The leakage rate of nitrogen through extremely small holes is less than half that of helium.

2) More space experience is available with nitrogen pressurization than with any other system (e.g. Ranger and Mariner).

The amount of weight increase resulting from the selection of nitrogen over helium is not considered as important as the reduction in the leakage rate. The self-contained system will permit modular packaging of the propulsion system.

The self-contained helium system was rejected because of the increased leakage rate.

Tank pressurization by means of a hydrazine gas generator is not considered as developed as the nitrogen system at this time.

The weight of the blowdown system was excessive due to the increased tank pressure. This system requires the installation of an undeveloped liquid regulator, if thrust level control is desired.

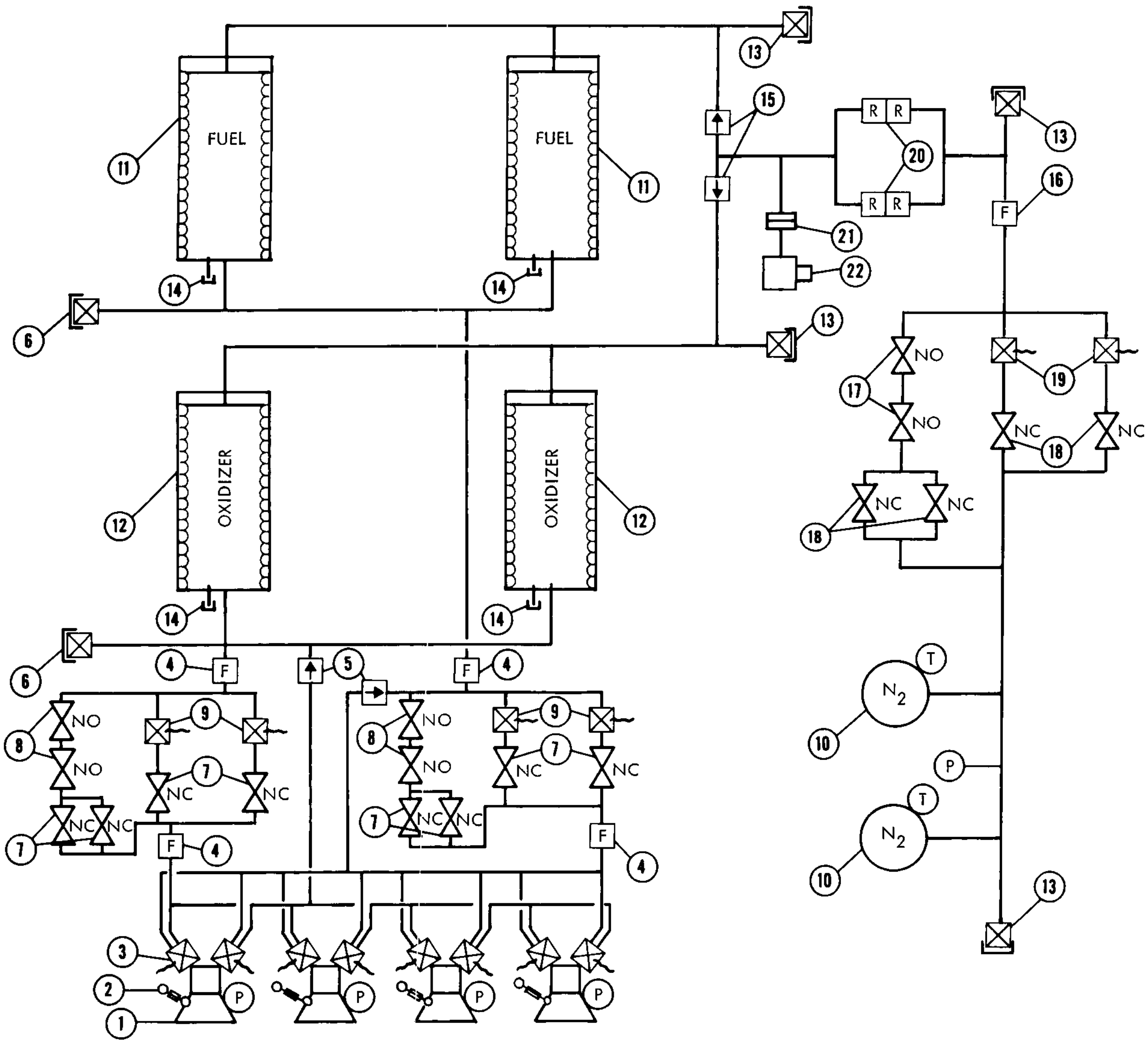
Storage of the tank pressurization gas in the reaction-control system tankage compromises the reliability of both the reaction-control and liquid propulsion systems. It also prevents the modular packaging of both systems.

4.2.2.6 Isolation Valving and Plumbing

Candidate Fluid Systems--The fluid systems considered for use in the bipropellant midcourse propulsion system are presented in Figures 4.2-14, 4.2-15, and 4.2-16. The three systems all include four cylindrical storage tanks using metal bellows as the expulsion device. A self-contained nitrogen-tank pressurization system is included in two spherical tanks for storage. The systems differ in the amount of redundancy provided in flow-control devices.

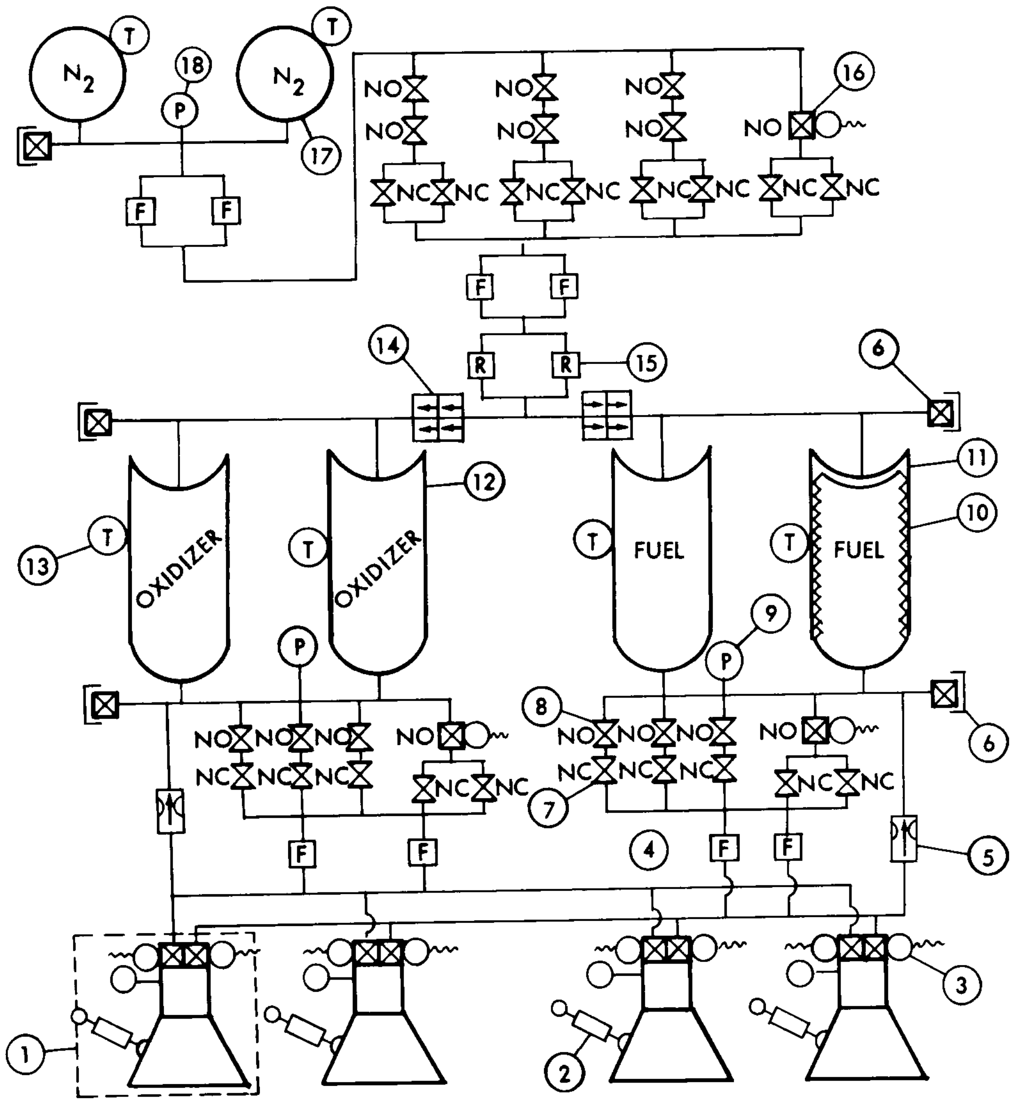
Competing Characteristics--The following characteristics were considered in the final selection of the preferred fluid system:

1) Reliability;
2) Zero leakage during long-term shutdown;



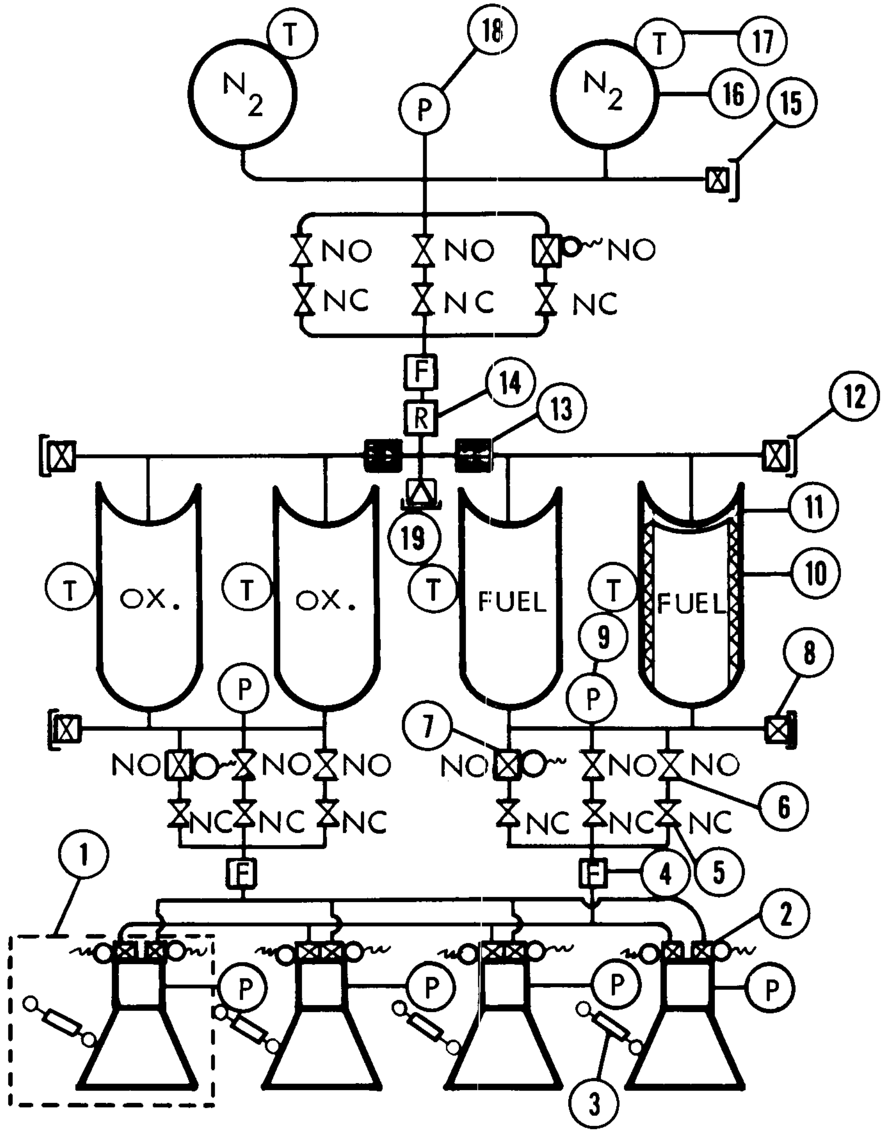
ITEM NO.	QTY	ITEM
22	1	RELIEF VALVE
21	1	BURST DISC
20	2	REGULATOR, DUAL, N_2 PRESS.
19	2	VALVE, N_2 SOL. LATCHING
18	4	VALVE, N.C. N_2 PRESSURE SQUIB OPERATED
17	2	VALVE, N.O. N_2 PRESSURE SQUIB OPERATED
16	1	FILTER, N_2 PRESSURE
15	2	VALVE, CHECK
14	4	VALVE, VENT PROPEL. FILLING
13	4	VALVE & CAP FILL/TEST
12	2	TANK, OXID BELLOWS EXPULSION
11	2	TANK, FUEL BELLOWS EXPULSION
10	2	TANK, N_2 PRESSURE
9	4	VALVE, PROPELLANT SOLENOID LATCHING
8	4	VALVE, N.O. PROPELLANT SQUIB OPER
7	8	VALVE, N.C. PROPELLANT SQUIB OPER
6	2	VALVE & CAP FILL/TEST
5	2	VALVE, THERMAL RELIEF
4	2	FILTER, PROPELLANT
3	8	VALVE, BIPROPELLANT SHUT-OFF SOL. OPERATED
2	8	GIMBAL ACTUATOR ASSY
1	4	ROCKET ENGINE ASSY BIPROPELLANT

Figure 4.2-14: Bipropellant Midcourse And Orbit Trim — Isolation Valving And Plumbing (System A)



ITEM NO	QTY	ITEM
18	1	TRANSDUCER, PRESSURE (HIGH)
17	2	TANK, STORAGE, NITROGEN
16	3	VALVE, SOLENOID
15	2	VALVE, PRESSURE REGULATOR
14	2	VALVE, QUAD CHECK
13	6	TRANSDUCER, TEMPERATURE
12	2	TANK, STORAGE, OXIDIZER
11	2	TANK, STORAGE, FUEL
10	4	BELLOWS, POSITIVE EXPULSION
9	2	TRANSDUCER, PRESSURE (LOW)
8	12	VALVE, SQUIB, NORMALLY OPEN
7	18	VALVE, SQUIB, NORMALLY CLOSED
6	5	VALVE, FILL & VENT
5	2	VALVE, THERMAL RELIEF
4	8	FILTER
3	4	VALVE, BIPROPELLANT SOLENOID ACTUATED
2	8	GIMBAL ACTUATOR ASSY
1	4	ENGINE, ROCKET ASSY

Figure 4.2-15: Bipropellant Midcourse And Orbit Trim — Isolation Valving and Plumbing (System B)



ITEM NO.	QTY	ITEM
19	1	VALVE, PRESSURE RELIEF
18	1	TRANSDUCER, PRESSURE (HIGH)
17	6	TRANSDUCER, TEMPERATURE
16	2	TANK, NITROGEN STORAGE
15	1	VALVE, FILL, GAS
14	1	VALVE PRESSURE REGULATOR
13	2	VALVE, QUAD CHECK
12	2	VALVE, VENT
11	4	TANK, PROPELLANT STORAGE
10	4	BELLOWS, POSITIVE EXPULSION
9	6	TRANSDUCER, PRESSURE (LOW)
8	2	VALVE, FILL, LIQUID
7	3	VALVE, SOLENOID, NORMALLY OPEN
6	6	VALVE, SQUIB, NORMALLY OPEN
5	9	VALVE, SQUIB, NORMALLY CLOSED
4	3	FILTER
3	8	ACTUATOR, GIMBAL ASSY
2	4	VALVE, BIPROPELLANT SOLENOID
1	4	ENGINE, ROCKET ASSY

Figure 4.2-16: Bipropellant Midcourse And Orbit Trim — Isolation Valving And

3) Absence of catastrophic failure modes;

4) Simplicity;

5) Weight.

Selection Rationale--Table 4.2-6 is a summary of the characteristics of the fluid systems considered. Based upon this analysis, System A, as shown on Figure 4.2-14, was selected as the preferred system for the following reasons:

1) Sufficient operating paths are provided to perform the required maneuvers, and redundant shutoff valves are installed in such a manner so that no single control device failure can cause catastrophic failure of the mission. This is considered to be very important.

2) Overall system reliability is high.

3) The system is maintained in a zero-leakage condition by closed squib valves during the time from initial charging until the second midcourse maneuver.

4) Filling of the fuel and pressurant tanks and checkout of the system can be accomplished easily.

5) The weight of the system is acceptable.

The preferred system (Figure 4.2-14) uses brazed or welded connectors between all fittings, components, and tubing. With careful fabrication and inspection techniques, an essentially zero-leakage system is provided. The filters installed in both the liquid and gaseous portions of the system will be of sufficient capacity to provide the required reliability. Nitrogen-pressure regulators will be of the type used on the Mariner program, to assure reliable flight-proven hardware.

Table 4.2-6 ISOLATION VALVING AND PLUMBING--BIPROPELLANT SYSTEM

	A	B	C
Number of Components	64	70	44
System Weight, lb	283.81	291.84	272.44
Total Number of Connections	237	274	153
System Reliability	0.99868	0.99893	0.998105
Redundant Components			
Pressure regulators	3	1	0
Filters	0	3	0
Relief valves	0	0	0
Squib valves	12	8	4
Solenoid valves	6	4	0
Check valves	0	6	6
Number of Shutoff Valves			
Squib	18	30	15
Solenoid	16	11	11
Manual	4	5	5
Loss of System Failure Modes	0	2	5

The system shown on Figure 4.2-15 was found to be the most reliable of those considered, but included an extra flow path that would have permitted two initial midcourse correction maneuvers. This was concluded to be unnecessary, and the increase in the number of components and system weight was not justified by the slight increase in reliability.

The fluid system presented in Figure 4.2-16 was attractive because of its simplicity, light weight, and minimum number of components. It was rejected, however, because a single failure of a flow-control device could cause catastrophic failure of the vehicle.

4.2.3 Liquid-System Selection

Description--Monopropellant and bipropellant systems using Earth-storable propellants were considered for the midcourse-correction and orbit-trim propulsion system.

Competing Characteristics--The liquid propulsion system selection was based on the following competing characteristics:

1) Reliability;
2) Simplicity;
3) Space-use experience;
4) Failure mode characteristics;
5) Development requirements;
6) Mission performance requirements;
7) Growth.

Selection Rationale--The monopropellant system is the simplest and most reliable system. It has fewer potential failure modes, including those of a hazardous character. Fewer development-program problem areas are associated with this system in applications involving prolonged exposure to space. Greater operating experience in deep-space missions has also accrued with monopropellant systems (Ranger, Mariner) than with bipropellant systems. A monopropellant system was selected.

4.3 SOLID/LIQUID THERMAL CONTROL

4.3.1 Propulsion Module Thermal Control

The selected thermal design controls propulsion-module temperatures independently of equipment bays, rather than thermally coupling to them. This choice is made so as not to widen the range of temperatures experienced in the equipment bays. It is justified in Volume A, Section 4.1.12. The major thermal requirements of the solid/liquid system are summarized in Table 4.3-1.

4.3.1.1 Candidate Approaches

The following thermal control approaches were considered:

1) Adjusting heat leak into and out of the module to a desired level, using both conventional louvers and electric heat for temperature control.
2) Adjusting heat leak into and out of the module to a desired level, using electric heat only for temperature control.
3) Adjusting heat leak into and out of the module to a desired level, using solar louvers for temperature control.

In each concept, insulation is used on the interior surface of the bus and around the solid motor and its nozzle. This provides thermal isolation of the propulsion module from the equipment bays, space, the Sun, and exhaust plume heating.

4.3.1.2 Competing Characteristics

Competing characteristics in the selection of the method of thermal control were:

1) Reliability;
2) Control margin;

Table 4.3-1: TEMPERATURE LIMITS OF PROPULSION MODULE SYSTEMS

System		Two Sigma Operational Limits -- °F		Qualifying Limits** °F		Basis
		Lower	Upper	Lower	Upper	
Solid Motor (Modified M^2)	Boost	21	99	-15	135	Vendor Data
	Transit	21	109	-15	145	Vendor Data
	Firing	21	99	-15	135	Vendor Data
Monopropellant System		75	131 Max	39	167 Max	Mariner Experience
Bipropellant System		40	100	4*	136	State of Art
Equipment Compartment (Reference)		50	80	24	116	Design Choice

* Heater required, freezes at 12°F

** JPL required: 36°F above and below 2σ operation

3) Technical risk;

4) Weight.

4.3.1.3 Selection Rationale and Discussion

A comparison of the three candidate approaches is given in Figure 4.3-1. With Approach (1), varying electric heat does not have an important effect on system design. With Approach (2), it is possible to use fixed louvers or other fixed-heat-leak designs. However, a minimum of 95 watts is required. Solar louvers, Approach (3), are seen to be inferior to conventional louvers for this design.

Approach (1) is therefore selected. Approach (3) shows no performance advantage, and is not yet space-proven. Approach (2) incurs large power penalties. Also, the overall reliability of Approach (2), approximately 0.988, is lower than the reliability of 0.9998 for Approach (1).

4.3.2 Bipropellant Thermal Control

The candidate bipropellant engines were reviewed. No serious thermal problems are apparent. Minor modifications may be necessary to adapt the engines to Voyager needs.

4.3.3 Monopropellant Thermal Control

The key monopropellant engine thermal problem is that of maintaining the catalyst bed temperature above 30°F. This problem occurs during misorientation when the engine is in the shadow.

4.3.3.1 Candidate Approaches

The following approaches were considered:

1) Engine nozzle exposed, with electric heater on catalyst bed;

2) Insulated engine nozzle;

3) Low-emissivity nozzle exterior.

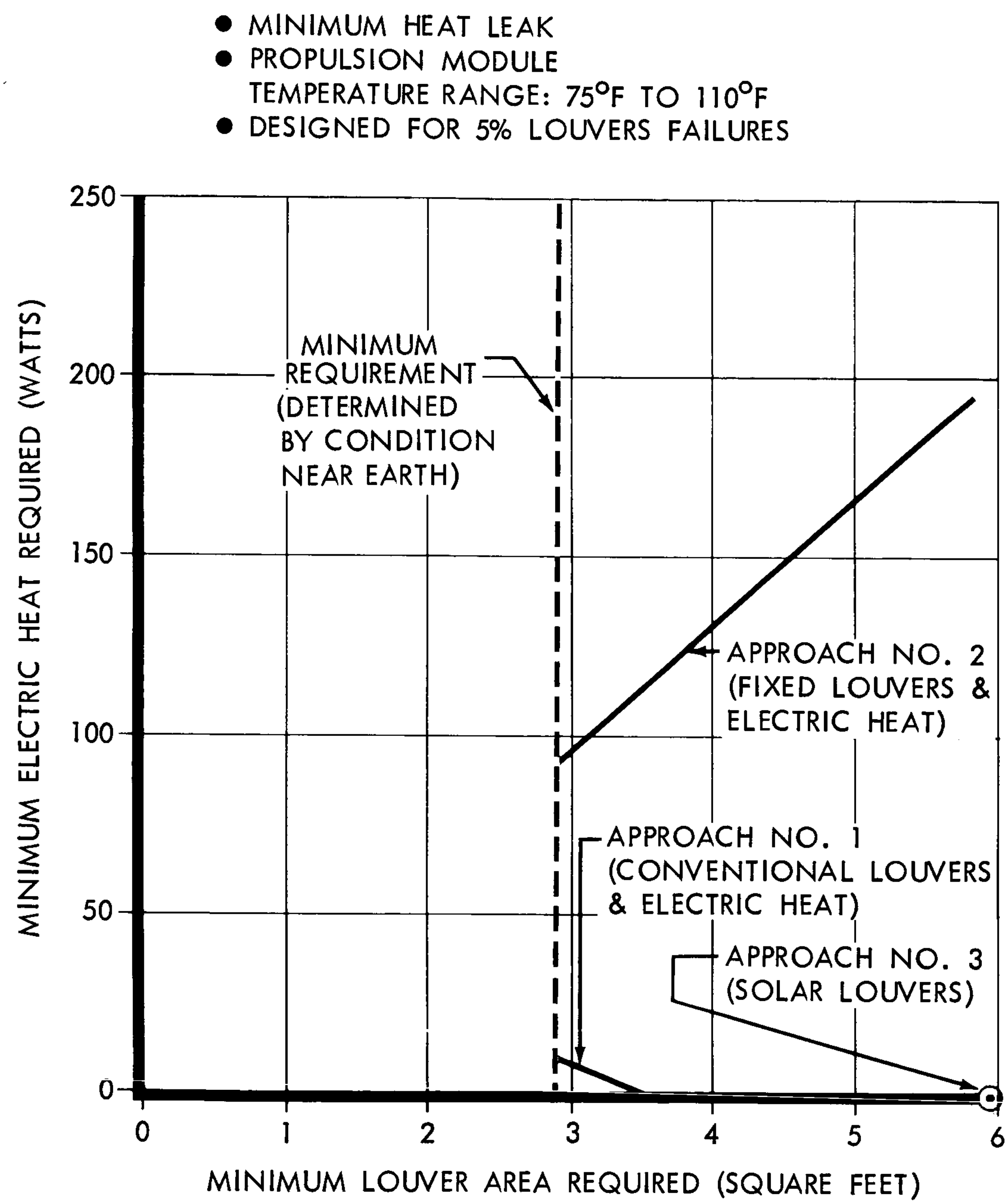


Figure 4.3-1: Solid/Liquid Units Propulsion Module Temperature Control Options

4.3.3.2 Competing Characteristics

Competing characteristics in the selection of the preferred approach were:

1) Reliability;

2) Interaction with propulsion module temperature control.

4.3.3.3 Selection Rationale and Discussion

Temperatures during misorientation are given in Figure 4.3-2 for 1) an insulated nozzle; 2) an uninsulated nozzle; and 3) an intermediate case of a low-emissivity nozzle exterior, which is equivalent to a partially insulated nozzle. The data show that the insulated nozzle maintains satisfactory temperatures. The low-emissivity nozzle is marginally satisfactory. The uninsulated nozzle drops below the lower design limit during off-Sun maneuvers. An electric heater is therefore required for the uninsulated nozzle, with a power consumption of 90 watts. The weight penalty for this power requirement compares unfavorably with an incremental weight penalty of approximately 3 pounds for the insulated nozzle.

An added consideration is the effect of the engines on the overall thermal balance of the propulsion module. Approach (1) has the least effect, since there is little temperature difference for heat transfer into or out of the propulsion module. Approach (2) results in less than 20 watts heat leak into the propulsion module, as shown in Figure 4.3-3.

Approach (1), control by use of insulation, is selected as the preferred method for engine temperature control. This passive approach has a higher reliability than heaters. The insulation is considered to have a smaller weight penalty than the power penalty of Approach (2). Although the use of electric heaters produces the lowest thermal interaction with the propulsion module, the interaction produced by the preferred approach is acceptable.

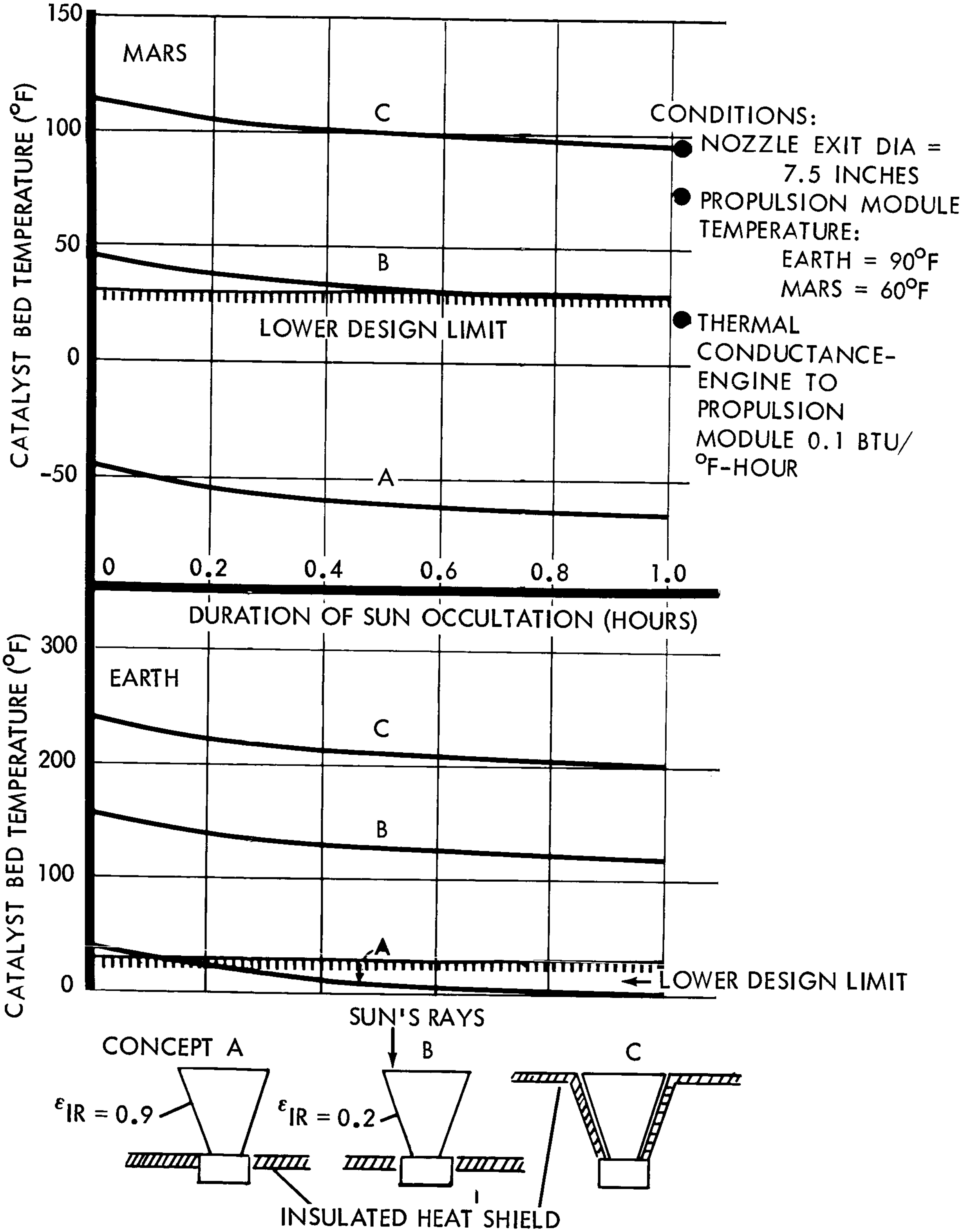


Figure 4.3-2: Monopropellant Engine Catalyst Bed Temperature During Occultation

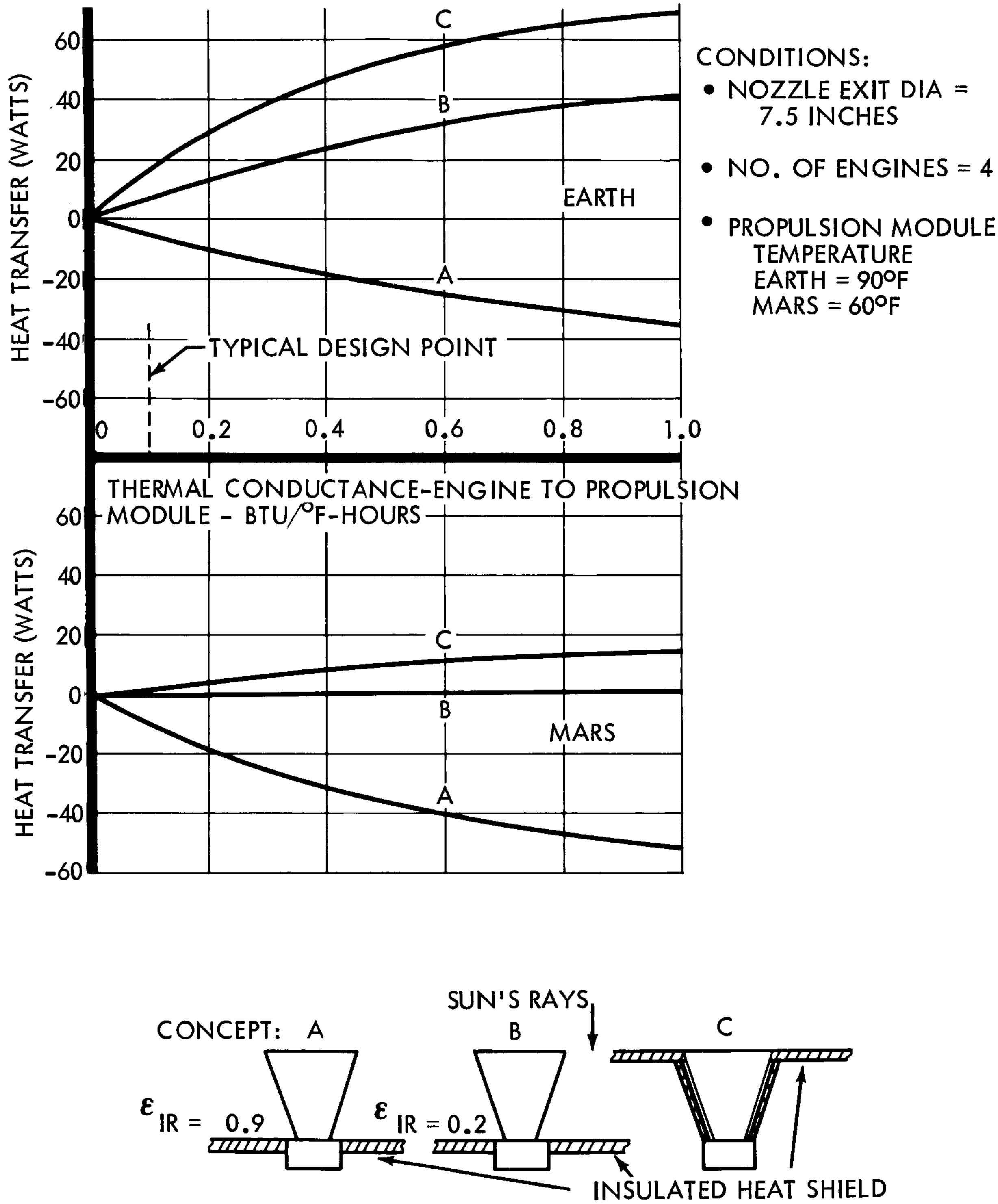


Figure 4.3-3: Heat Transfer To Propulsion Module Through Monopropellant Engines

4.4 SOLID/LIQUID LENGTH TRADES

The effects of solid/liquid propulsion system length on spacecraft subsystem design were considered. Comparison of representative configurations with realistic length options is shown in Figure 4.4-1.

Configuration A (945-8055) was selected because it allows the use of an existing solid motor without compromising the following:

1) Overall length (The solid motor is the overall length-determining factor for the optimized 1971 configuration, but only by about 3 inches. For the 1975 mission, the hydrazine tanks are the propulsion system length limiting factor.)
2) Thermal control
3) Antenna size
4) Canopus tracker field-of-view

The selected design is conservative as it allows for a length margin.

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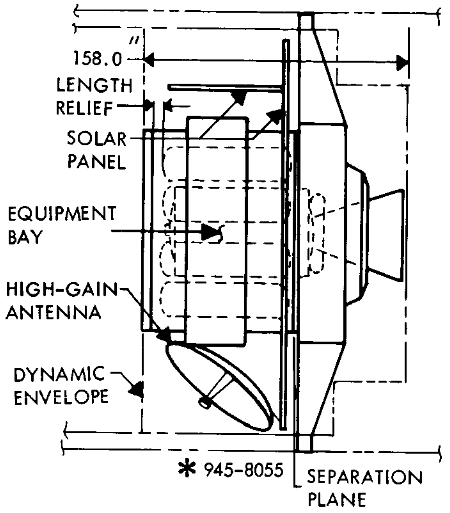
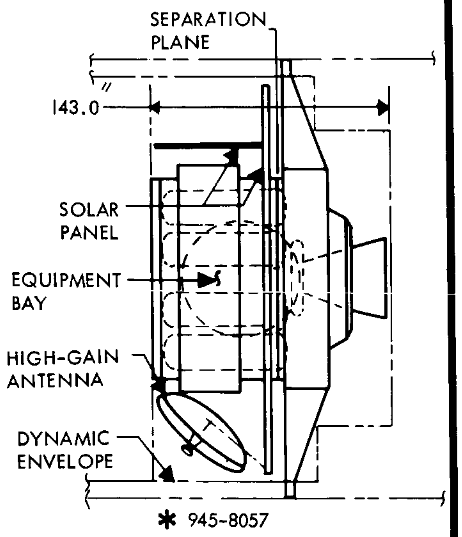
	GENERAL CONFIGURATION	GUIDANCE AND CONTROL
CONFIGURATION A  	• Adaptability to fixed solar panel area: Good • Stowage volume utilization: Good • Thermal control views: Good • C.G. control: Good • Separation characteristics: Good • Configuration 945-8055 is preferred spacecraft design utilizing a modified Minuteman solid-propellant motor (see Section 3.10 of Volume A). Length relief added for design conservatism and growth allowance.	• Reaction control subsystem weight: 165 lbs for '71 mission • Thrust vector control (TVC). Midcourse, (jet vanes) Cant angle -- 14° Cant angle propellant penalty -- 3% TVC pointing error -- 0.15 degrees Orbit insertion: Secondary injection Thrust vector freon weight (inc residual) -- 203 lbs TVC pointing error -- 0.37 degrees • View factors: Solar panels positioned at aft end of spacecraft to accommodate desired Canopus tracker field of view.
CONFIGURATION B  *BOEING DESIGNATION	• Adaptability to fixed solar panel area: Good • Stowage volume utilization: Fair • Thermal control views: Fair • C.G. control: Good • Separation characteristics: Good • Configuration 945-8057 adapts the same spacecraft arrangement to a spherical solid motor of the same performance as the modified Minuteman 945-8055. It shortens the spacecraft the maximum possible. The high volumetric efficiency of solid-motor designs tends to eliminate the motor as a factor affecting spacecraft length (Configuration 945-8057 is shown without length relief to emphasize this fact). Length-limiting factors are: Solar panels, Hydrazine tankage, Antenna size, Thermal design requirements, Guidance and control view factors.	• Reaction control subsystem weight: 157.4 for '71 mission • Thrust vector control (TVC). Midcourse, jet vanes Cant angle -- 31° Cant angle propellant penalty -- 17% TVC pointing error -- increased due to shorter length Orbit insertion: secondary injection Thrust vector freon weight Increased due to shorter length TVC pointing error -- increased due to shorter length • View factors: Field of view of redundant Canopus tracker compromised by undeployed antenna.

COMMUNICATIONS	THERMAL CONTROL	STRUCTURAL FACTORS
• Maximum antenna: 6.5-foot diameter. • Antenna arm storage: Simple	• Temperature control acceptable. A long nozzle is used on the solid for added conservatism on plume heating. This causes the solid motor to be the limiting factor on bus length when the propulsion system is optimized for 1971-1973, but only by approximately 3 inches. When the propulsion system is optimized for 1975, general configuration considerations limit bus length rather than the propulsion system.	• Weight: Fair • Spacecraft load distribution: Fair • Launch shroud loading: Continuous • Shroud weight penalty: 11 lbs per inch • Spacecraft structure penalty: 2 lbs per inch
• Maximum antenna: 5.5-foot diameter. • Maximum ellipsoid 5.5 foot by 7.5 foot. • Antenna arm stowage: complex	• Temperature control is acceptable. For 1975 mission, the short bus length causes a limited view of louvers to space, resulting in reduced but acceptable performance.	• Weight: Good • Spacecraft load distribution: Fair • Launch shroud loading: Continuous • Shroud weight penalty: 11 lbs per inch • Spacecraft structure penalty: 2 lbs per inch

Figure 4.4-1: Spacecraft - Length Effects On Subsystem Design (Solid-Liquid Propulsion System)

5.0 SOLID/LIQUID SYSTEM OPTIMIZED FOR 1975 & 1977

The optimized solid/liquid system concept does not change in 1975 and 1977. Because of the increased Planetary Vehicle weight, the ratio of solid-to-liquid propellant decreases as follows:

	1971 & 1973	1975 & 1977
Solid Propellant Required, lb	9839	9045
Liquid Propellant Required, lb	2495	3190

This results in an optimum 1975 insertion motor that is 6.0 inches shorter than the 1971 motor.

6.0 LEM DESCENT PROPULSION SYSTEM

In adapting the LEM descent propulsion system to the Voyager mission, only mandatory modifications, or those where a significant improvement in reliability or performance is realized, were implemented.

6.1 MANDATORY MODIFICATIONS (Excluding Thermal Control)

6.1.1 Pressurization Gas Storage

The LEM descent propulsion system uses cryogenic-stored helium for propellant tank pressurization. This is changed to ambient-temperature stored helium because of the long mission time.

6.1.2 Landing Gear - deleted

6.2 PROPELLANT SETTLING

Candidates--The following three methods were considered: main-tank surface-tension screens, bipropellant settling rockets with positive expulsion, and monopropellant settling rockets with positive expulsion.

Competing Characteristics--The primary competing characteristics are reliability, technical risk, and weight.

Selection Rationale and Discussion--Main tank screens and bipropellant settling rockets were rejected for the following reasons:

1) Main tank surface-tension screens--Because of its relative newness, this method involves considerable technical risk. Compatibility with Voyager mission profile would be difficult and expensive to prove.
2) Bipropellant settling rockets--The system considered is similar to the one discussed in Section 4.2.1. Performance improvement over a

monopropellant system is not considered worth the reliability decrement.

The preferred method of main-tankage propellant settling is by monopropellant rockets with their own positive expulsion. This system has the highest reliablity and minimum technical risk.

6.3 THRUST VECTOR CONTROL

Candidate Systems--Three alternate schemes are considered:

1) Pulsed-jet thrust vector control with LEM engine fixed;
2) Pulsed-jet thrust vector control with LEM engine gimbaled using existing actuator as a trim device;
3) LEM engine only with high-performance gimbal actuator.

A comparative sketch of the above systems is shown in Figure 6.3-1.

Competing Characteristics--The following competing characteristics, in decreasing order of priority, were considered in selecting the preferred TVC system.

1) Reliability;
2) Availability;
3) Minimum impact on length under the shroud;
4) Weight;
5) Pointing accuracy;
6) Growth.

Selection Rationale and Discussion--Pulsed-jet control with LEM engine trim and gimbaled LEM engine with a high performance actuator were rejected for the following reasons:

1) Pulsed jet with LEM engine trim--To obtain the benefit of using the slow-speed actuator of the LEM engine as a trim device to minimize

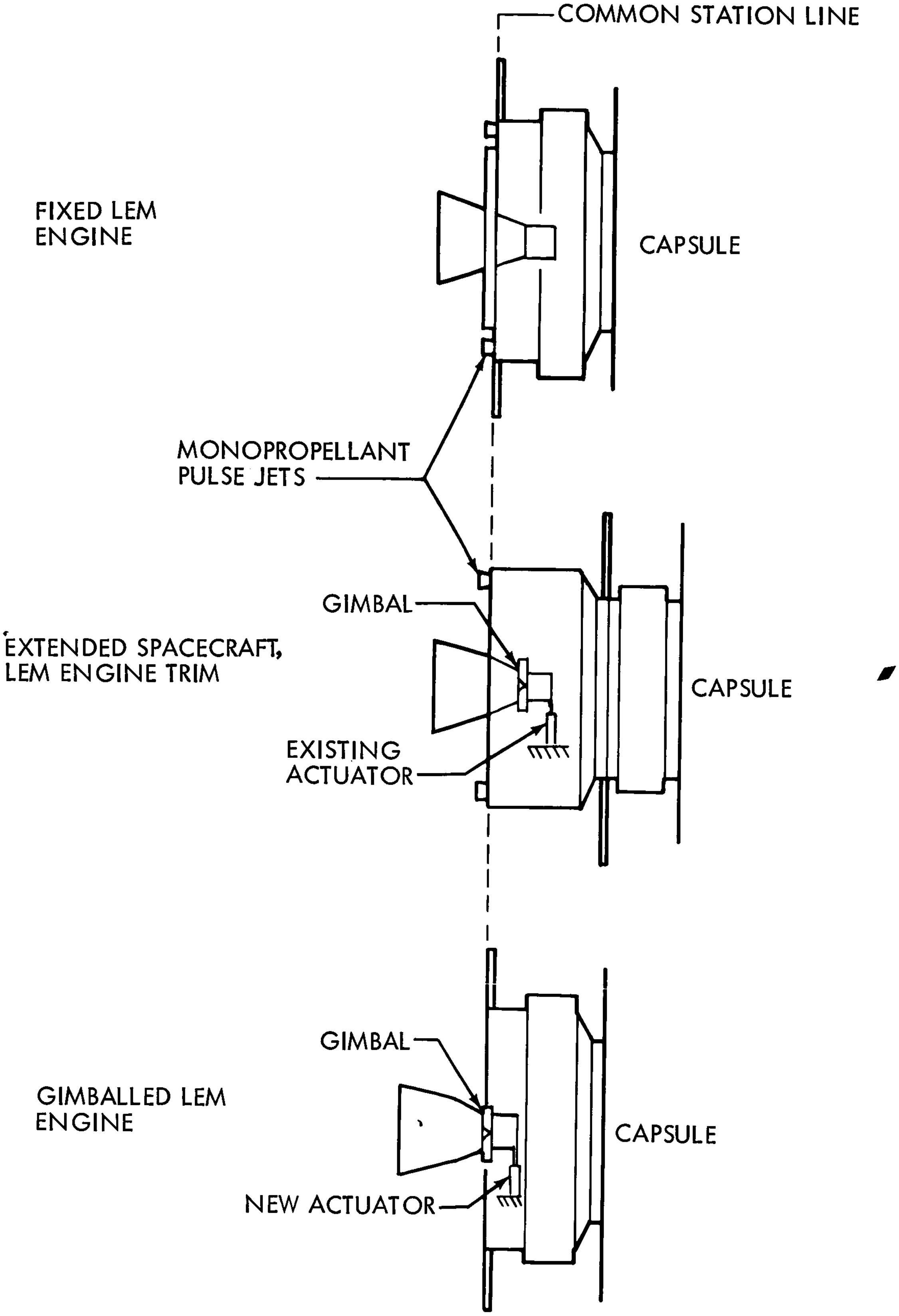


Figure 6.3-1: LEM Thrust-Vector-Control Alternate Configurations

c.g. offset errors, an unduly long spacecraft is necessary to obtain a suitable c.g. to engine gimbal point distance. The alternative of allowing the engine gimbal to bottom and accept a partially reduced c.g. offset error is rejected because of possible instabilities. Consequently, the normal gimbaled LEM engine with pulse jets is rejected because of inability to provide satisfactory control without increasing spacecraft length, with its attendant increase in weight and length under the shroud.

2) High-Performance Gimbal Actuator Control--This configuration is also sensitive to the distance from c.g. to engine gimbal. A feasible mechanization requires the extension of the LEM engine about 20 inches aft of its nominal position. The attendant difficulties of providing a high-performance, presumably hydraulic, actuator at this location--together with tail-wag-dog and pointing-error problems--preclude the selection of this type of system. Moreover, this system represents a significant structural modification to the LEM module, obviating many of the advantages of using an existing stage.

The pulse-jet system, with LEM engine fixed, provides the only feasible system that can be accommodated within the limits of a short-length spacecraft without extensive system redesign. This TVC approach was recommended as an adequate backup to secondary injection in Volume B of the Final Report for Task A. The system uses four 100-pound monopropellant thrusters mounted symmetrically about the LEM engine at a moment arm of 6.57 feet. Four additional engines could provide cooperative redundancy. It is questionable, however, whether the added complexity results in a realistic reliability gain.

The TVC engines also provide the necessary tank settling prior to the firing of the LEM engine. Under these conditions, it is necessary to provide attitude control with the engines in the "normally on" mode. This mode is switched at main engine ignition to a "normally off" mode to minimize TVC monopropellant requirements. A block diagram of the system is shown in Figure 6.3-2.

Typical ranges of requirements and performance parameters for a "normally off" jet system are shown in Table 6.3-1. The LEM descent engine thrust level is assumed to be 10,500 pounds for orbit insertion and 1050 pounds for all other maneuver modes.

6.3.1 Selected System Performance in Terms of Competing Characteristics

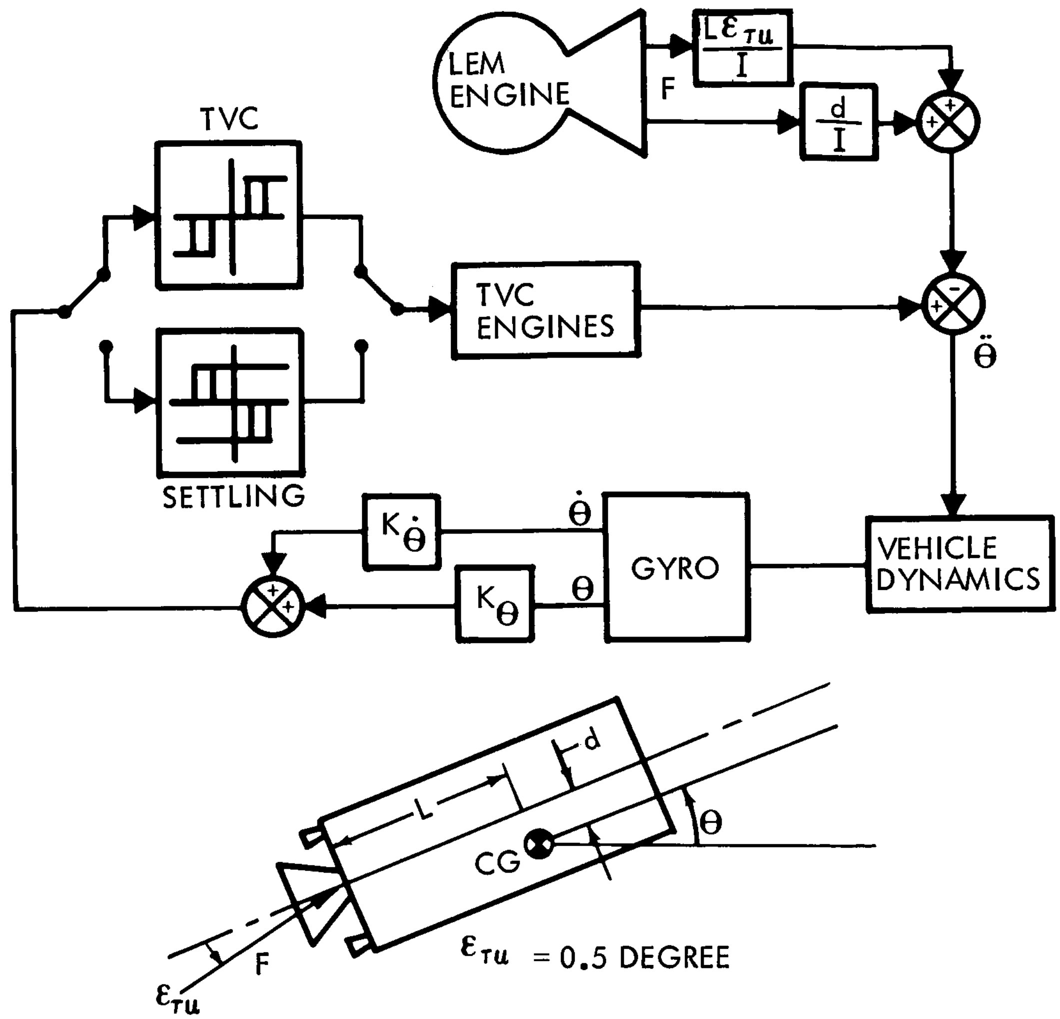
Reliability--Reliability of monopropellant engines with the new spontaneous decomposition catalyst in pulsed operation is not fully characterized, but is considered acceptable.

Availability--Monopropellant engines in the 100-pound-force range are state-of-the-art hardware.

Minimum Impact on Spacecraft Length--The selected system results in minimum spacecraft length.

Weight--System weight attributable to TVC is reflected in additional tankage and monopropellant required over the system weight necessary for LEM engine tank settling.

Pointing Accuracy--Pointing accuracy is a function of the selected system deadband and thrust misalignment. Deadbands of the order of ± 0.5 degree are considered practical.



d INCLUDES 0.125 INCH FOR LEM ENGINE ALIGNMENT

	1971 MISSION 2000-POUND CAPSULE		1975 MISSION 10,000-POUND CAPSULE	
	L ins	d ins	L ins	d ins
CAPSULE OFF				
MIDCOURSE	-1	0.393	0	0.515
ORBIT INSERT.	-1	0.393	0	0.515
ORBIT TRIM	3	0.925	6	0.835
CAPSULE ON				
MIDCOURSE	9	0.365	44	0.345
ORBIT INSERT.	9	0.365	44	0.345
ORBIT TRIM	28	0.495	76	0.475

Figure 6.3-2: LEM With Pulsed Engines Thrust-Vector-Control

Table 6.3-1: LEM PULSED-JET THRUST VECTOR-CONTROL PERFORMANCE PARAMETERS

	1971 MISSION (2000-LB CAPSULE)							1975 MISSION (10,000-LB CAPSULE)				
	CAPSULE OFF			CAPSULE ON				CAPSULE OFF		CAPSULE ON		
	MID-COURSE	ORBIT INSERTION	ORBIT TRIM	MID-COURSE	ORBIT INSERTION	ORBIT TRIM	MID COURSE	ORBIT INSERTION	ORBIT TRIM	MID-COURSE	ORBIT INSERTION	ORBIT TRIM
TOTAL IMPULSE, LB-SEC	171.5	14,200	74.4	231	15,650	79.6	238	17,900	79	51.8	22,400	208
PULSE WIDTH, SEC	0.058	0.115	0.031	0.085	0.194	0.058	0.061	0.180	0.037	0.251	4.25	0.174
PROPELLANT WEIGHT,LB	0.75	61.8	0.34	1.01	68.1	0.35	1.06	77.8	0.36	2.25	95.4	0.91
DUTY CYCLE, %	5.24	52.5	12.6	5.9	59.3	10.1	6.87	68.7	11.8	9.7	97.0	12.35
NUMBER OF PULSES PER ENGINE	30	1228	25	26	805	14	39	993	22	21	52	12

Assumptions: Worst c.g. Offset

± 0.5 Degree Deadband

Thrust levels selected for TVC can only marginally accommodate the 1975 and 1977 mission maximum disturbance torques. A slight increase in thrust level may be required for these later missions.

6.4 LEM DESCENT PROPULSION MODULE THERMAL CONTROL

The LEM propulsion module normally operates between 40° and 100°F. If this temperature range were permitted on Voyager, then application of temperature margins (±36°F) for system FAT would cause propellant freezing. Therefore, the system temperature limits are narrowed to between 50° and 100°F.

The selected design controls the temperatures independently of the spacecraft bus. This choice narrows the temperature range experienced in the equipment bays. It is justified in Volume A, Section 4.3.5.

Necessary LEM propulsion design changes for thermal-control purposes are:

1) Changes required to survive soakback heating (i.e., restart successfully) after engine firing. In the existing design, maximum allowable temperatures are exceeded locally to the point where vapor forms and propellant decomposition may occur. (This is not a problem in the Apollo application because restart capability after the prolonged engine firing is not required.)

2) Changes necessary to prevent engine shut-off valve overheating because of engine solar heating. (This is not a problem in the Apollo application where, unlike Voyager, the engine nozzle does not normally face the Sun).

Candidate Approaches--The following alternate approaches are available for LEM propulsion thermal control.

1) Adjusting heat leak in and out of the module to the desired level by using both conventional louvers and electric heat.
2) Adjusting heat leak in and out of the module to the desired level, Using only electric heat for temperature control.
3) Adjusting heat leak in and out of the module to the desired level, Using solar louvers for temperature control.

Competing Characteristics--The following competing characteristics were considered, in decreasing priority:

1) Technical risk,
2) Control margin,
3) Reliability,
4) Weight.

Selection Rationale and Discussion--Heat-leaks for the LEM descent propulsion system are large because the propulsion module structure is used as the primary structure of the bus. Many insulation penetrations are required. Consequently, relatively large amounts of louver area or electric heat are needed to maintain adequate thermal control.

A comparison of the three candidate approaches is given in Figure 6.4-1, based on the spacecraft layout shown in Figure 2.3-1. With Approach (1), the use of a small amount of electric heat to assist in thermal control results in a large saving in required louver area. With Approach (2), it is possible to adapt fixed louvers or other fixed heat-leak designs; however, a minimum of 210 watts is required. Conventional louvers require less area than solar louvers if power is available for control. Solar louvers (Approach (3)) are better when power is not available.

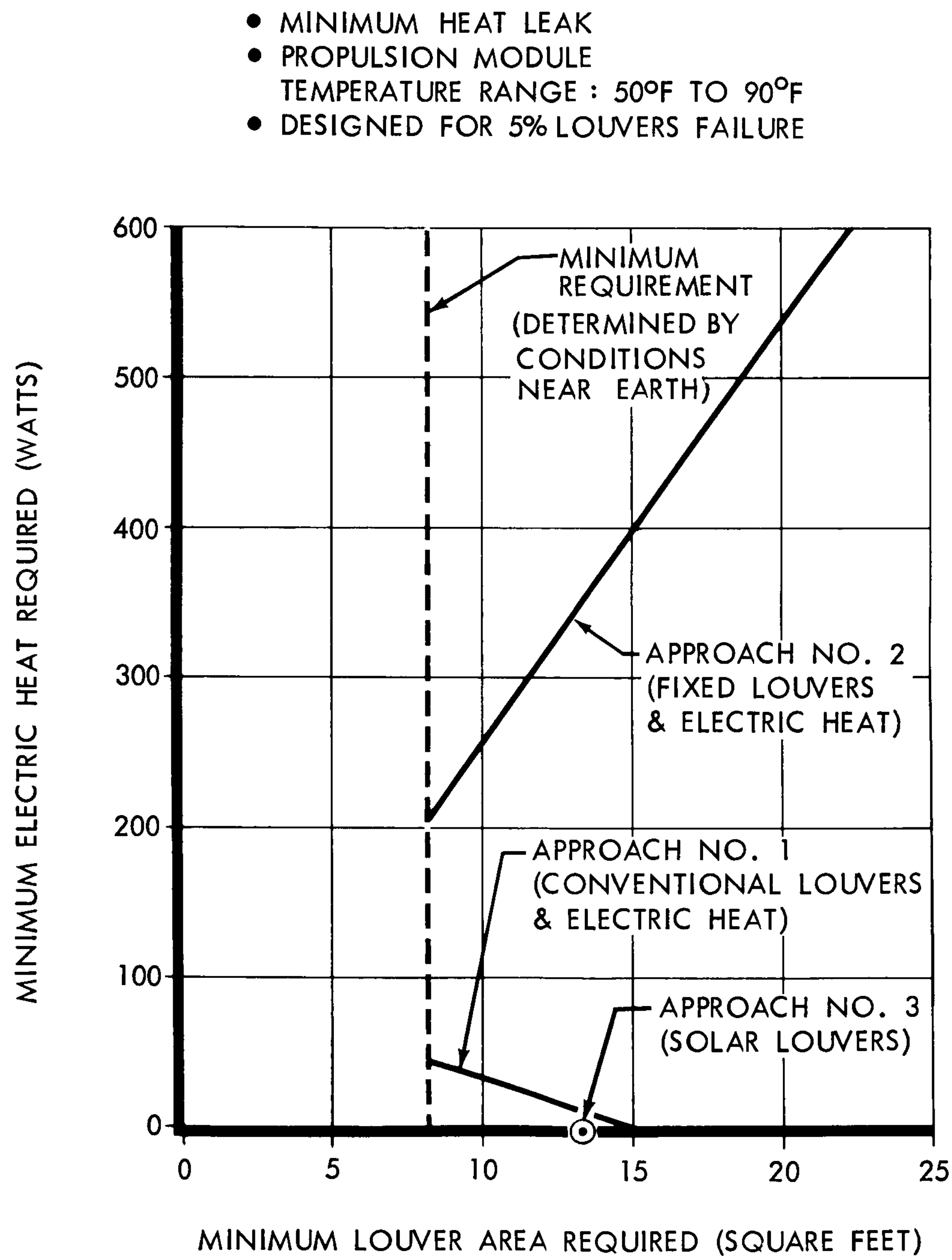


Figure 6.4-1: LEM Descent Propulsion Module Temperature Control Options

The preferred approach for temperature control of the LEM propulsion module is Approach (1)--a combination of louvers and electric heat, for the following reasons:

1) A surplus of at least 100 watts of electric power is still available prior to final orbit trim. This is adequate for the needs of Approach (1).

2) This approach has the least technical risk as it is least sensitivie to uncertainties in heat leaks which are large for the LEM propulsion system.

Approach (3) (solar louvers) is rejected because it is not a space-proven method. Approach (2) requires 210 watts. A surplus as high as 210 watts cannot reasonably be anticipated. The additional 110 watts for Approach (2) incurs a large power penalty.

6.5 LEM DESCENT LENGTH TRADES

The effects of LEM descent propulsion system length on spacecraft subsystem design were considered. Comparison of two configurations representing feasible lengths is shown in Figure 6.5-1.

Configuration B (945-8012A) was selected because:

1) Shroud length is shortest, and

2) Launch shroud loading is continuous.

The smaller antenna size associated with this configuration is not a constraining factor, and the increased thrust vector control system weight is more than compensated for by shroud length reduction.

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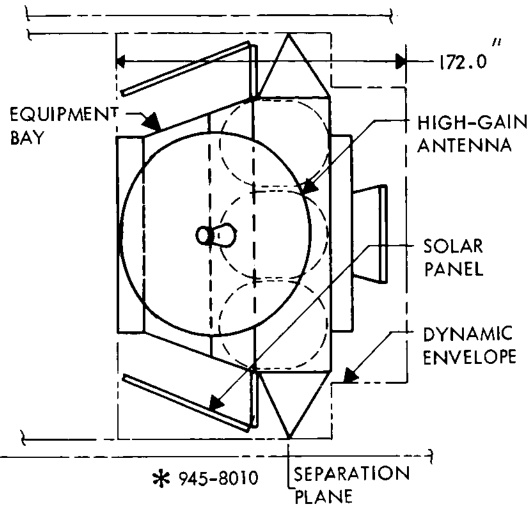
CONFIGURATION A

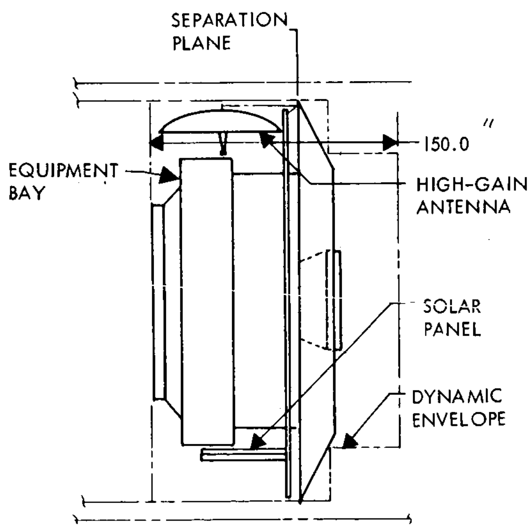


* 945-8010

GENERAL CONFIGURATION

- Adaptability to fixed solar-panel area: Poor
- C. G. control: Poor
- Stowage volume utilization: Good
- Launch separation: Poor
- The Lunar Excursion Module descent stage application, Configuration 945-8010, is characterized by a forward adapter section in which the spacecraft equipment is mounted. Also featured is spacecraft-mounted adapter structure which applies point loading to the shroud. These features tend to produce good stowage and deployment capability, but high spacecraft length and weight.

CONFIGURATION B



* 945-8012

* BOEING DESIGNATION

- Adaptability to fixed solar-panel area: Poor
- C. G. control: Very poor
- Stowage volume utilization: Poor
- Launch separation: Good
- The Lunar Excursion Module descent stage application (Configuration 945-8012A) provides a minimum-length spacecraft with an aft-placed continuous-loading Launch Vehicle Adapter and equipment mounted around the octagonal periphery of the basic LEM structure. The result is poor utilization of the dynamic envelope and poor panel- and antenna-stowage characteristics.

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GUIDANCE AND CONTROL	COMMUNICATIONS
• Reaction control subsystem weight: 223.3 lbs • Thrust vector control (TVC). Midcourse and orbit insertion, (Pulse jets plus LEM trim) TVC Pointing Error--0.5 deg Pulsed Engine Propt.--2.5 lbs No. of pulses--57 • System uses LEM descent engine to trim for c.g. offset errors. Pulsed jets used because LEM actuator rate, ± 0.4 deg per sec max, is inadequate.	• Maximum antenna diameter: 10 feet
• Reaction control subsystem weight: 196.5 lbs • Thrust vector control (TVC). Midcourse and orbit insertion, (pulse jets) TVC Pointing Error--0.5 deg Pulsed Engine Propt.-70.8 lbs No. of Pulses--896 • Pulsed system required because of c.g. excursions throughout spacecraft mission.	• Maximum antenna diameter: 6.5 feet

Figure 6.5-1:

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THERMAL CONTROL	STRUCTURAL FACTORS
• Long length permits locating equipment bays away from solar-panel thermal influence and permits excellent temperature control of equipment bays. Control of propulsion module is acceptable. For 1975 mission, the conical section causes a limited view of louvers to space, resulting in reduced but acceptable performance.	• Weight: Poor • Spacecraft load distribution: Fair • Launch shroud loading: Discontinuous • Shroud weight penalty: 11 lbs per inch • Spacecraft structure penalty: 2 lbs per inch
• Temperature control is acceptable.	• Weight: Fair • Spacecraft load distribution: Fair • Launch shroud loading: Continuous • Shroud weight penalty: 11 lbs per inch • Spacecraft structure penalty: LEM structure is utilized

Spacecraft - Length Effects On Subsystem Design (LEM Descent Propulsion System)

3

7.0 TITAN III-C TRANSTAGE

In adapting transtage to the Voyager mission, the only modifications implemented were either absolutely required, or else resulted in significant improvements in probability of mission success.

7.1 MANDATORY MODIFICATIONS

7.1.1 Engine Propellant Valves

The existing thrust chamber valves are not compatible with Voyager mission leakage constraints. This is corrected by adding low-leakage pre-valves.

7.1.2 Plumbing Joints

All plumbing joints are brazed or welded for the Voyager mission application.

7.1.3 Meteoroid Shield

This is required to reduce the probability of damage to acceptable levels.

7.1.4 Tank Gage

Tank gages have been increased to comply with the required 2.2 safety factor.

7.2 SHORTENED TRANSTAGE

The transtage tankage has greater capacity than that required for the Voyager application. A 20-inch reduction in transtage length is feasible with minimal change. Further shortening requires redesign of the tank sway brace structure, which is considered a major modification. The significant result of shortening transtage length is a reduction of shroud length and booster aerodynamic loads.

7.3 PROPELLANT SETTLING AND PRESSURANT STORAGE

7.3.1 Propellant Settling

The preferred method of main-tankage propellant settling is monopropellant rockets with their own positive expulsion. Selection logic is the same as that applied in Section 6.2.

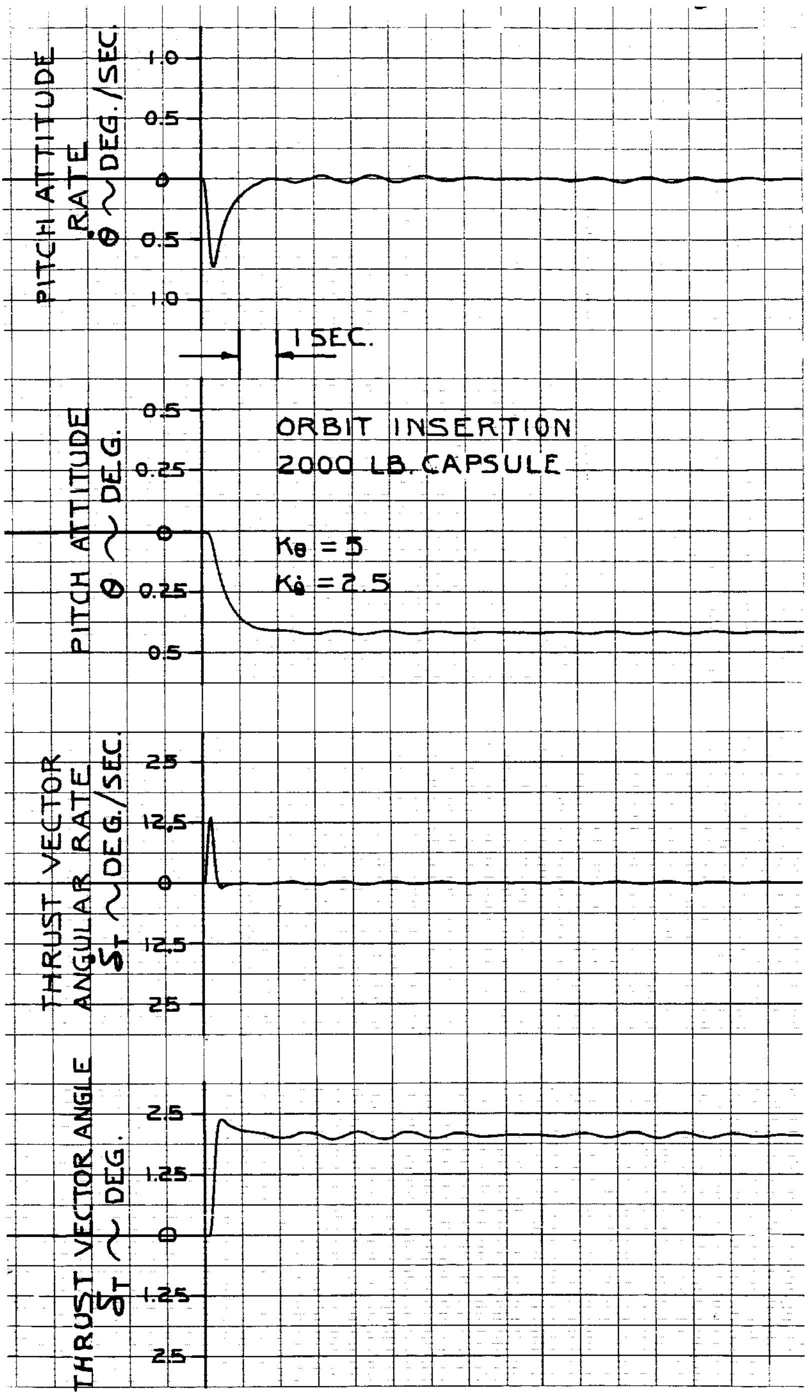
7.3.2 Pressurant Storage

Offloading the transtage tanks provides a considerable increase in the ullage volume. This ullage volume can be utilized for pressurant storage. By storing eleven pounds of helium, out of a total of 45.6 pounds, in the main tanks ullage volume, helium tankage weight saving of 150 pounds is realized. It is assumed that the reliability degradation resulting from this change is acceptable.

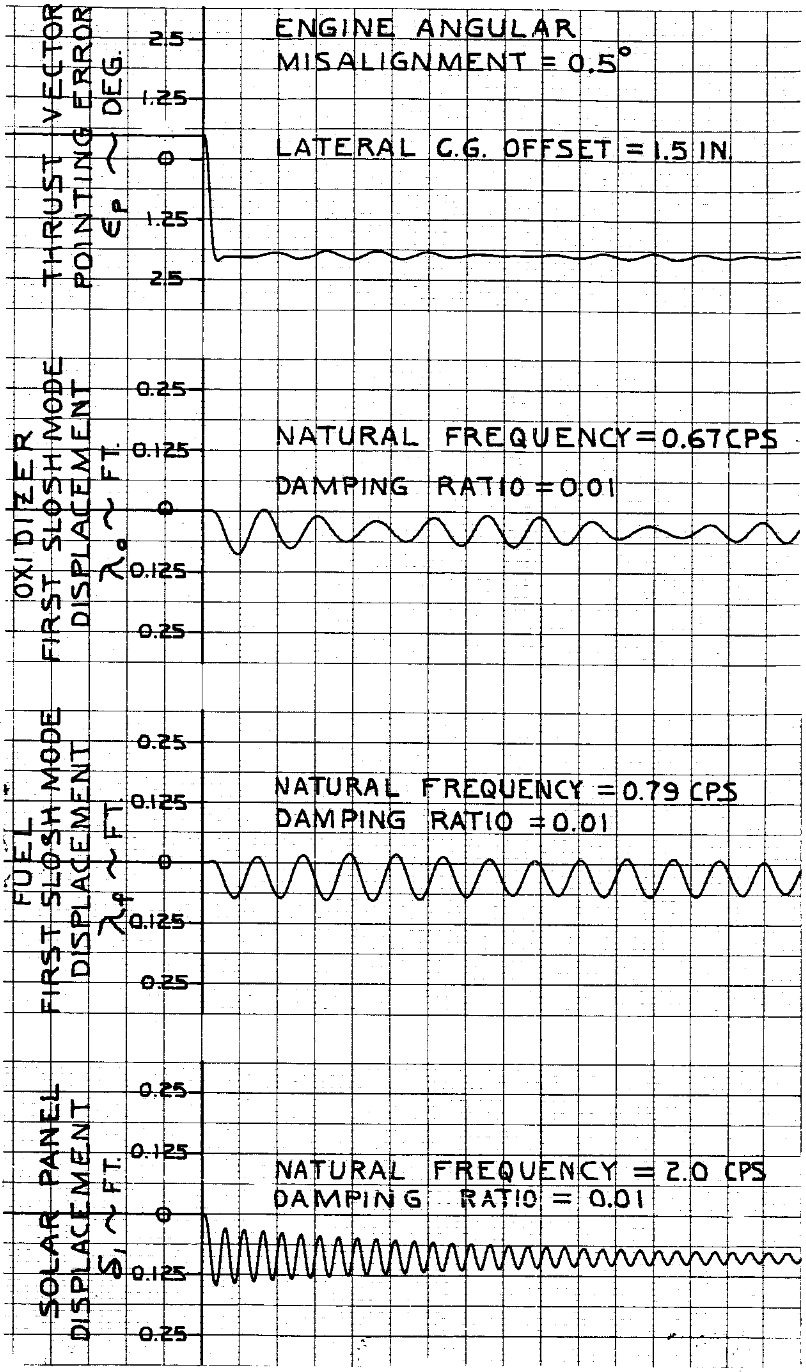
7.4 THRUST VECTOR CONTROL

An analysis of the Titan III transtage TVC system has been made to characterize its application to Voyager.

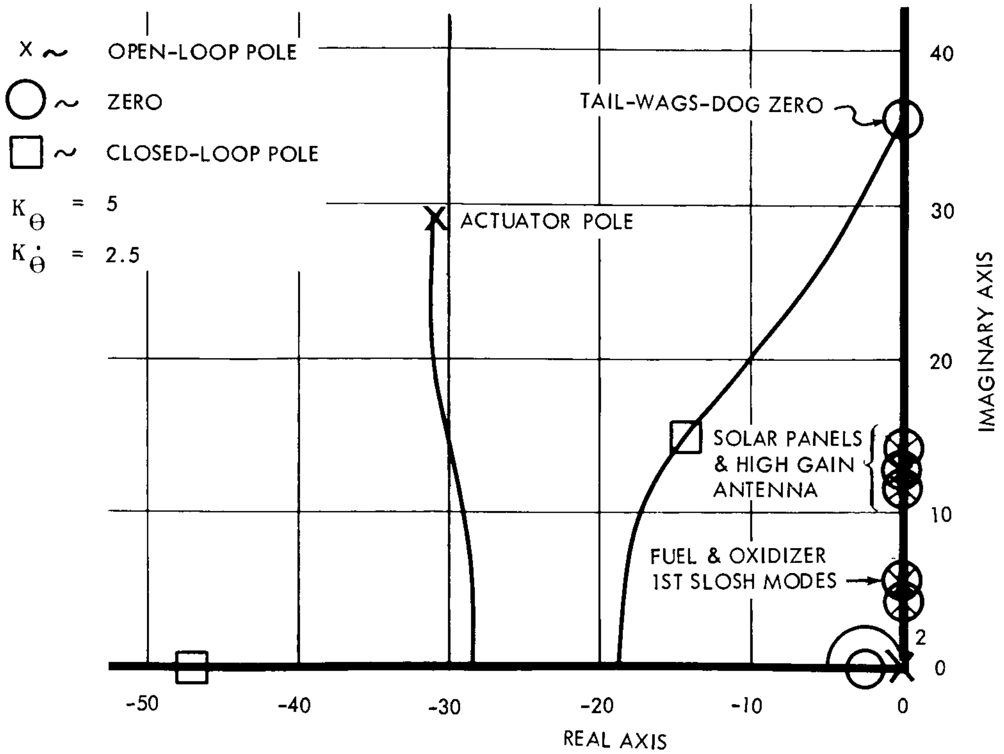
From a dynamic standpoint, the Titan III transtage thrust vector-control system is feasible for Voyager application. An example of the time transient at engine ignition is shown in Figure 7.4-1, along with a functional block diagram of the simulated system. A root locus plot of the closed-loop poles of the system is also shown. For purposes of comparison between propulsion systems, perfect rate and position feedback signals were assumed. For the example shown, the fuel and oxidizer slosh modes were stable. However, an extensive study is necessary to review slosh mode stability at all fluid levels in the tanks. The effect of structural coupling and tail-wags-dog (engine inertial reaction) did not prove significant.



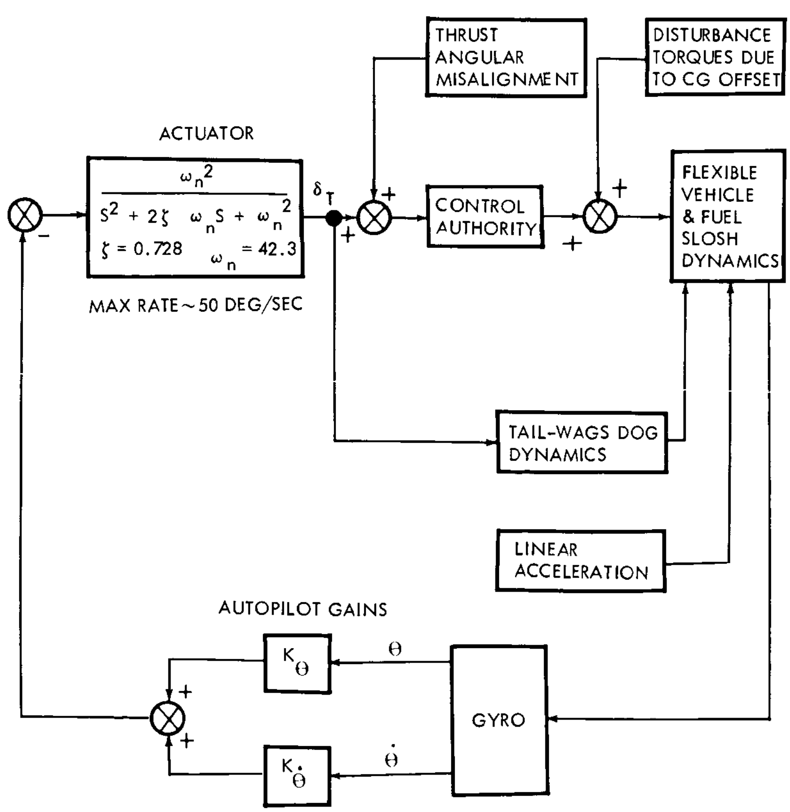
(C) START BURN TIME RES



PONSE — ORBIT INSERTION



(A) ROOT LOCUS PLOT — TVC SYSTEM CLOSED-LOOP POLES



(B) AUTOPILOT FUNCTIONAL BLOCK DIAGRAM

Figure 7.4-1: Titan III Transtage Thrust Vector Control

A minimum burn time of approximately one second is probably required to reduce the transients in spacecraft attitude rates following engine ignition. This is necessary to prevent gyro position output saturation during recovery with the low-level reaction-control system. The propellant settling engines are provided with jet-vane thrust vector control as they may be used alone to implement a minimum correction in velocity.* Without TVC, tolerances in angular alignment, thrust levels, and c.g. offset cause spacecraft angular rates at burnout that are too large for effective recovery with the low-level reaction-control system.

From a static standpoint, the transtage system is less desirable than the solid/liquid system because of thrust vector pointing accuracy. Due to the unsymmetric placement and uneven loading of the tanks, the Voyager spacecraft c.g. will shift through a lateral range of 2.13 inches as propellants are consumed. A thrust vector pointing error will therefore occur unless the spacecraft attitude is purposely biased, prior to firing, to take this into account. However, tolerances in predicting the lateral c.g. position will vary as high as ± 0.54 inch (orbit trim without capsule) due to errors in propellant mixture ratio. A minimum c.g.-to-trunnion distance of approximately 56 inches, and a thrust angular alignment tolerance of ± 0.5 degree were assumed. Practical limits on thrust vector pointing accuracy are then on the order of 0.70 degree with realistic autopilot gains. This is within the approximate 1-degree error budgeted to the TVC system for Voyager.

It is concluded that the Titan III transtage TVC system is satisfactory for Voyager applications.

*Note: Transtage settling subsystem operation with engine-out appears feasible by canting the Hydrazine engines and relocating them closer to the roll axis. This increases overall transtage reliability from 0.9907 to 0.9947. This does not alter the preferred design selection.

7.5 TITAN III-C TRANSTAGE PROPULSION MODULE THERMAL CONTROL

The selected thermal design controls temperatures independent of bus equipment bays. This choice is made so as not to widen the range of temperatures experienced in the equipment bays. It is justified in Volume A, Section 4.3.5.

Transtage temperature control limits are normally 45° to 90°F. No specific deficiencies were noted in transtage that would create a serious thermal problem. However, insulation must be added for each of the candidate approaches discussed below. The system-design operating-temperature limits of 45° to 90°F must be narrowed to 50° to 90°F to prevent propellant freezing when operating to FAT limits.

Competing Characteristics

1) Technical risk
2) Control margin
3) Reliability
4) Weight

Selection Rationale and Discussion--A comparison of the three candidate approaches is given in Figure 7.5-1, based on the spacecraft layout that is shown in Figure 2.4-1. With Approach (1), the use of a small amount of electric heat to assist in control results in small louver areas. With Approach (2), fixed louvers or other fixed heat-leak designs are feasible. However, a minimum of 165 watts is required. Conventional louvers (Approach (1)) require less area than solar louvers (Approach 3) if power is available. Otherwise, solar louvers are preferred. Solar louvers are better suited to the Titan III-C transtage propulsion system than to either the LEM descent or solid/liquid system; Figure 7.5-2

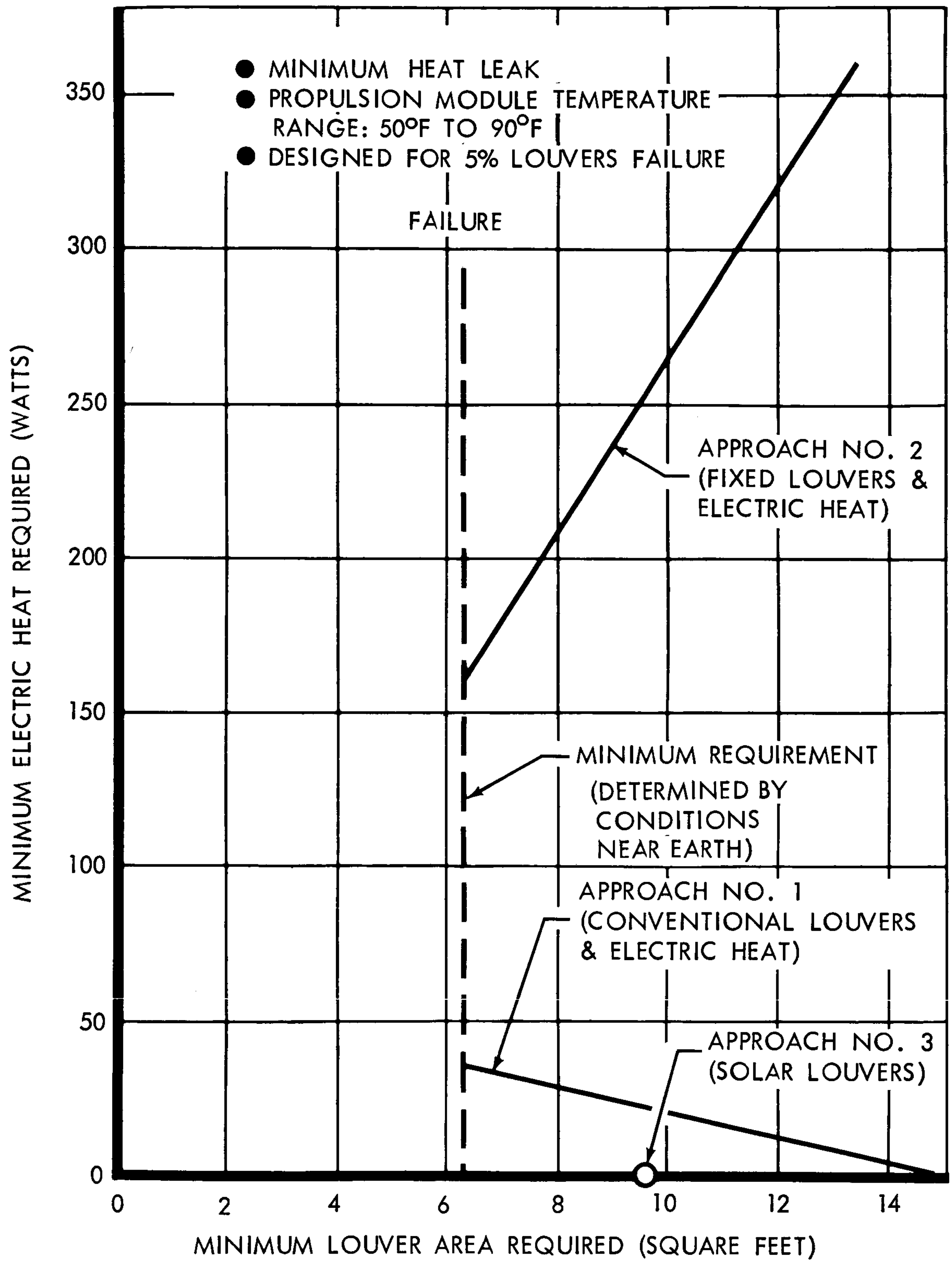
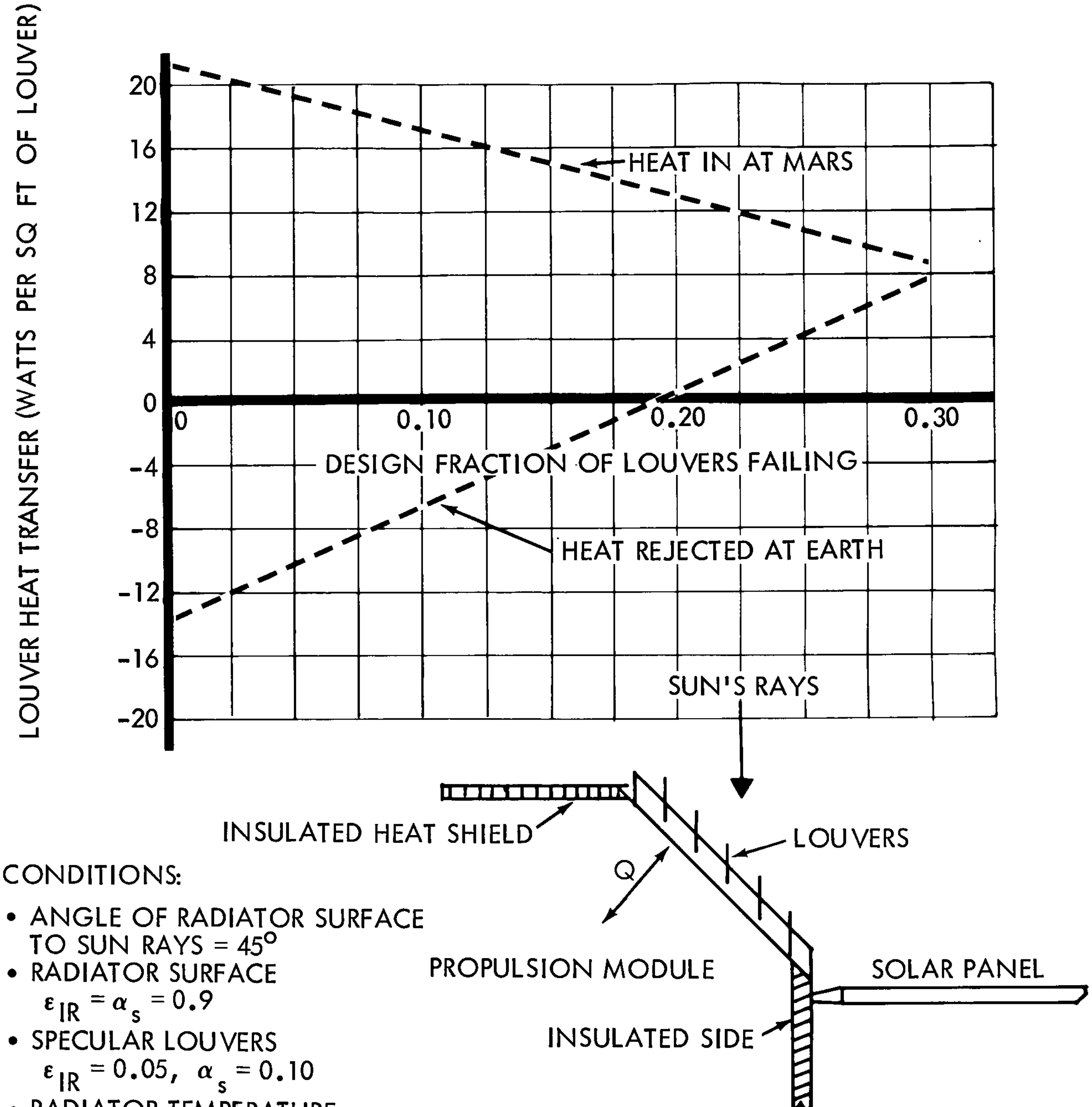


Figure 7.5-1: Titan III-C Transtage Propulsion Module Temperature Control Options



CONDITIONS:

- ANGLE OF RADIATOR SURFACE TO SUN RAYS = 45°
- RADIATOR SURFACE $\varepsilon_{IR} = \alpha_s = 0.9$
- SPECULAR LOUVERS $\varepsilon_{IR} = 0.05, \ \alpha_s = 0.10$
- RADIATOR TEMPERATURE
  EARTH = 80°F
  MARS = 60°F
- LOUVERS FAIL IN WORST POSITION

Figure 7.5-2: Transtage Propulsion Module Heat Transfer Through Sun-Facing Louvers shows the performance characteristics of solar louvers and illustrates their capability to either add or reject heat.

The preferred approach for temperature control of transtage is Approach (1), a combination of conventional louvers and electric heat. The basis for this selection is minimum technical risk due to uncertainties in heat losses for this design. Approach (3) is rejected because it is not a space-proven concept. Approach (2) is rejected because design margins on heat losses result in high power requirements. A surplus of at least 100 watts of electric power is available prior to final orbit trim, after which time the propulsion module need no longer function. This is adequate for the needs of Approach (1), but not for those of Approach (2).

7.6 TRANSTAGE LENGTH TRADES

The effects of transtage propulsion system length on spacecraft subsystem design were considered. Comparison of two representative configurations of differing lengths is shown in Figure 7.6-1. Configuration B (945-8029) was selected as it resulted in the shortest length under the shroud without compromising spacecraft subsystem design.

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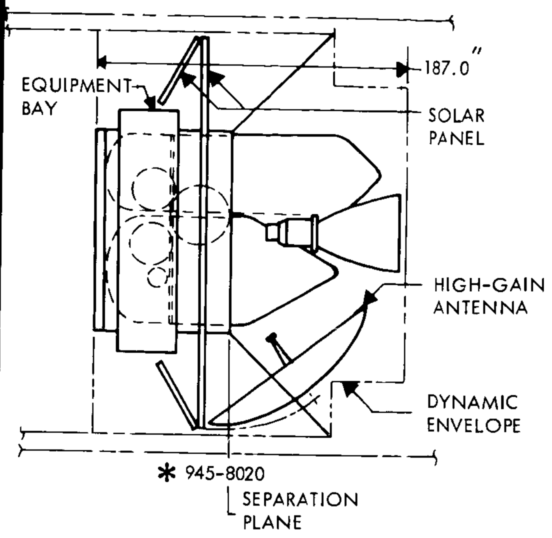
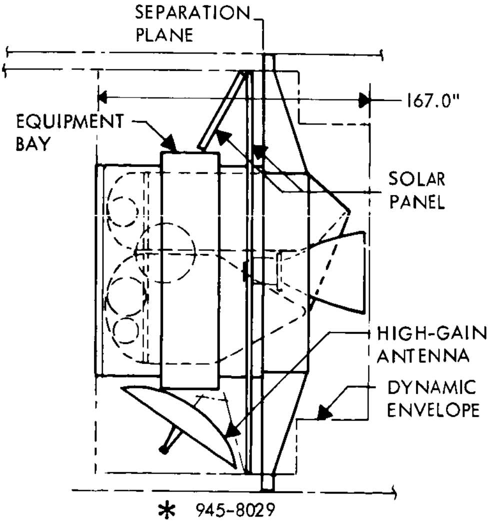
CONFIGURATION A



CONFIGURATION B



* BOEING DESIGNATION

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GENERAL CONFIGURATION	GUIDANCE CONTROL
• Adaptability to fixed panel area: Good • Stowage volume utilization: Fair • C.G. Control: Fair • Thermal control views: Fair • Shroud separation characteristics: Poor • The unmodified transtage application (Configuration 945-8020) provides for direct attachment of 120-inch transtage structural ring to aft end of primary spacecraft structure. This places solar panels and antenna hinge at midlength region of spacecraft, which tends to compromise thermal view factors, stowage volumes, and adapter design. Spacecraft length and weight are undesirably high.	• Reaction control subsystem weight: 195.7 lbs. • Thrust vector control (TVC). (Gimbaled engine) midcourse and orbit insertion • TVC pointing error: 1.9 degrees for 1.5 inches C.G. offset • C.G. radial offset varies 2.15" due asymmetric tankage. • Resulting pointing error effects must be trimmed out requiring more complex Autopilot mechanization.
• Adaptability to fixed panel area: Good • Stowage volume utilization: Good • C.G. control: Fair • Thermal control views: Good • Separation characteristics: Good • The modified transtage application (Configuration 945-8029) features transtage propellant tanks that have been shortened 20 inches by reducing forward cylindrical portion. Design provides for sliding the transtage 120-inch diameter module inside the primary spacecraft cylinder, thus optimizing solar-panel position, stowage characteristics, and adapter design.	• Reaction control subsystem weight: 191.5 lbs. • Thrust vector control (TVC) (Gimbaled engine) midcourse and orbit insertion • TVC pointing error: 2.2^{o} for 1.5-inch C.G. offset. • C.G. radial offset is again dominating feature of pointing errors. Pointing errors increased because of shorter length.

COMMUNICATIONS	THERMAL CONTROL	STRUCTURAL FACTORS
• Maximum antenna diameter: 10 feet	• Temperature control acceptable. For 1975 mission, with larger capsule, the placement of solar panels and equipment bays causes a limited view of louvers to space which results in reduced but acceptable performance.	• Weignt: Poor • Spacecraft load distribution: Good • Launch shroud loading: Discontinuous • Shroud weight penalty: 11 lbs per in. • Spacecraft structure penalty: 2 lbs. per inch.
• Maximum antenna diameter: 10 feet	• Temperature control acceptable. For 1975 mission, good view of louvers to space gives good temperature control.	• Weight: Fair • Spacecraft load distribution: Fair • Launch shroud loading: Continuous • Shroud weight penalty: 11 lbs per in. • Spacecraft structure penalty: 2 lbs per inch.

Figure 7.6-1: Spacecraft - Length Effects On Subsystem Design (Transtage Propulsion System)

8.0 PREFERRED DESIGN ASSESSMENT

The recommended preferred design concept is similar to that proposed in Task A. The major difference is the possibility of modifying an existing solid motor as opposed to developing a new one.

Potential solid motor problem areas considered in Task A were:

1) The inherent lack of versatility of a solid-propellant motor;
2) Two-phase plume flow with high radiosity;
3) Sterilization;
4) Swirl torques;
5) Space storability.

Adapting the Minuteman Wing VI motor to meet Voyager requirements introduces hardware problems in the following motor areas: (1) TVC components, (2) Motor liner, (3) Motor chamber pressure, (4) nozzle extension.

Reassessment of the Task A problem areas and assessment of the Minuteman motor problem areas are given below.

Versatility--Both new and modified solid motors are unable to terminate thrust. The effects are aggravated by vehicle weight variations at orbit insertion. This is caused by propellant consumption during mid-course (for trajectory corrections) which cannot be determined apriori.

The preferred design is sized for the 1975 and 1977 missions. The larger planetary vehicles for these missions result in a larger hydrazine sub-system; consequently, sufficient monopropellant is on board in 1971 to provide for orbit insertion vernier. This vernier capability, coupled with off-periapsis insertion and B-vector adjustments, provides the preferred design with a versatility equivalent to that of a pure liquid stage.

<u>Exhaust Plume</u>--Concern was expressed in Task A over excessive solar panel heating from the exhaust of the solid motor. Sufficient test data are available from the Minuteman program to define this heating with considerable confidence, and it was determined that extending the nozzle exit cone 15 inches provides adequate solar panel thermal margin.

<u>Sterilization</u>--Current data indicate that solid-motor sterilization is not required. If motor sterilization were still a requirement, considerable motor modification would be required. The selected motor cannot be heat-soak sterilized after assembly. A possible acceptable substitute would be motor assembly using only heat-soaked and decontaminated components or ingredients.

<u>Swirl Torques</u>--Motor-induced roll moments are difficult to predict analytically. Without a flight-test vehicle, they are also difficult to determine experimentally, as evidenced during Surveyor motor-roll tests at AEDC. Data are available from 12 Minuteman flights, however, which adequately characterize induced roll torques of the Minuteman motor.

<u>Space Storage</u>--Sealing the propellant from space by a diaphragm at the throat improves the suitability of the Minuteman motor for Voyager application since it is then stored under partial atmospheric conditions. Its present silo storage life is predicted to be 10 years. Preliminary propellant vacuum exposure testing has indicated that the selected propellant is able to withstand the high vacuum environment with acceptable degradation. It is felt that additional testing will verify the space storage compatibility of a semisealed Minuteman motor.

<u>TVC Components</u>--Adaption of most existing TVC components is considered a straightforward engineering problem. One exception is the freon

bladder and tank. Its present capacity is 80 pounds greater than is required for the Voyager application. Because of its good reliability record, redesign is not desirable. However, its capability to survive boost loads in the offloaded condition is unknown. Dynamic testing will resolve this problem.

Motor Liner--Propellant removal results in lowering chamber pressure and increasing burn time over the present Wing VI design. Thermal analysis of the modified motor indicates that the unmodified liner experiences less total heating than in the original Minuteman design.

Motor Chamber Pressure--Removal of a 36-inch long section of the Minuteman chamber reduces the average chamber pressure from * psia to 255 psia. Burn-time chamber pressure reduces to 60 psia. Firings of motors containing 100 pounds of aluminized composite propellants show a consistent small increase in combustion efficiency when the operating pressure is reduced from 1000 psia to 250 psia. This phenomenon is expected to continue well below 250 psia. The motor manufacturer indicates that no degradation of the delivered vacuum specific impulse of the current Minuteman motor is anticipated as a result of operating chamber pressures down to 50 psia.

Nozzle Extension--An exit cone extension of 15 inches is required to insure reduction of plume radiation to the solar panels to acceptable levels. This extension is compatible with the existing nozzle housing design and aft flange attachment. Attachment bending moment due to secondary fluid injection will be considerably reduced from the Minuteman application as shown by the following tabulation:

*See D2-82709-10 Classified Supplement - Reference Page 26

PARAMETER	APPLICATION MINUTEMAN	VOYAGER
Thrust During Maximum TVC Requirements, lb	55,000	44,000
Maximum Deflection, degrees	4.3	2.0
Moment Arm, inches	18	23
Maximum Moment, in-lb	72,700	35,300

In Task A, it was concluded that the selection of the Hydrazine monopropellant subsystem, employing the newly developed Shell 405 Spontaneous Decomposition Catalyst, did not result in significant development problems. Available test data obtained at JPL and elsewhere indicate that Hydrazine engines employing the Shell Catalyst can experience both rough combustion, and pressure spikes shortly after ignition. These problems can be eliminated through reactor and injector redesign if required, during the development phase.